IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re Application of:	:	
Wolfgang BILLINGER, et al.	:	
Application No. 10/053,666	:	TC/Art Unit: 3644
Filed: January 24, 2002	:	Examiner: Tien Quang Dinh
For: DEVICE FOR CONSTRUCTING MOVABLE PARTS WITH STRUCTURAL ELEMENTS OF AIRPLANES OR THE LIKE	:	Atty Docket: P67552US0

APPEAL BRIEF UNDER 37 C.F.R. § 41.37

MAIL STOP APPEAL BRIEF - PATENTS
Commissioner for Patents
PO Box 1450
Alexandria, VA 22313-1450

Sir:

Further to the Notice of Appeal filed November 9, 2009, Appellant submits the following Brief on Appeal.

A request for a three-month extension of time with appropriate fee is being filed concurrently herewith. In the event that any additional fees are required for consideration of this paper and any papers associated therewith, please charge such fees to our Deposit Account No. 06-1358.

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I. REAL PARTY IN INTEREST

The real party in interest of the subject matter of this appeal is Fischer Advanced Composite Components AG, which has its headquarters located at Fischerstrasse 9, Ried im Innkreis, Austria, A-4910, by virtue of the assignment recorded at Reel/Frame 012536/0302 in this application on January 24, 2002.

II. RELATED APPEALS AND INTERFERENCES

Appellant and Appellant's representative are not aware of any other appeals or interferences which will directly affect or be directly affected by or have a bearing on the decision of the Board of Patent Appeals and Interferences ("Board") in this appeal.

III. STATUS OF CLAIMS

The appealed claims are claims 15, 19-28, 30 and 32-36 which are currently pending in this application. Claims 15, 19, 21-23, 26, 27, 30 and 32-36 stand rejected as allegedly being unpatentable under 35 U.S.C. §103(a) over U.S. Patent No. 6,234,423 to Hirahara et al. ("Hirahara") in view of U.S. Patent No. 5,224,670 to Padden. Claim 20 stands rejected under 35 U.S.C. §103(a) as allegedly being unpatentable under Hirahara as modified by Padden in view of U.S. Patent No. 3,102,559 to Koppelman et al. ("Koppelman"). Copies of Hirahara and Padden are enclosed in the Evidence Appendix. A copy of the claims on appeal appears in the attached Claims Appendix.

Rejected Claims: 15, 19-23, 26, 27, 30 and 32-36

Allowed Claims: None.

Withdrawn Claims: 24, 25 and 28.

Objected to Claims: None.

Canceled Claims: 1-14, 16-18, 29 and 31.

IV. STATUS OF AMENDMENTS

Appellant filed a Request for Reconsideration (without claim amendment) under 37 C.F.R. § 1.116 ("the Request") on October 7, 2009. In the Request, Appellant responded to the Examiner's rejections by discussing why the combination of references cited by the Examiner did not render Applicant's claims prima facie obvious.

Appellant received an Advisory Action mailed October 16, 2009, indicating that the Request was considered but did not place the application into condition for allowance.

Appellant filed a Notice of Appeal on November 9, 2009. No amendments were or have been presented to the claims after the final rejection mailed May 7, 2009.

V. SUMMARY OF CLAIMED SUBJECT MATTER

Claims 15, 27 and 30 are the independent claims on appeal and are set forth in the Claims Appendix.

Appellant's invention as set forth in claim 15 is directed to a connecting device used in an aircraft to connect a movable part (2, 3, 4) of the aircraft with a structural component (1) of the aircraft (page 1, lines 3-5). This overall structure is illustrated in Figure 1 of the Appellant's application which is reproduced below.

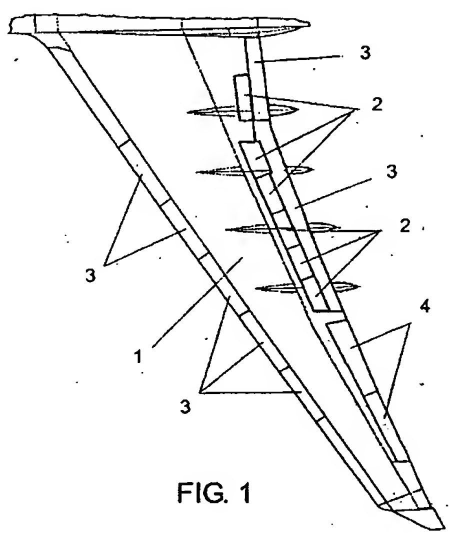

FIG. 1

The connecting device includes at least one fitting (5) having a movable part mounting structure (30, 40, 50) and a structural component connecting part (32, 42, 52) so as to be configured to connect the movable part with the structural component (page 7, second and third paragraphs, and page 7, last

paragraph, as amended by Amendment filed January 29, 2009; hereinafter "the January 2009 Amendment"). Such a fitting is illustrated in Figures 6 and 7 of Appellant's application which are reproduced below.

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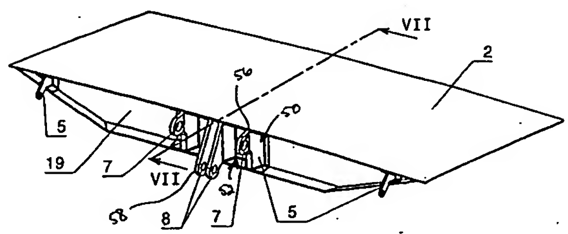


Fig. 6

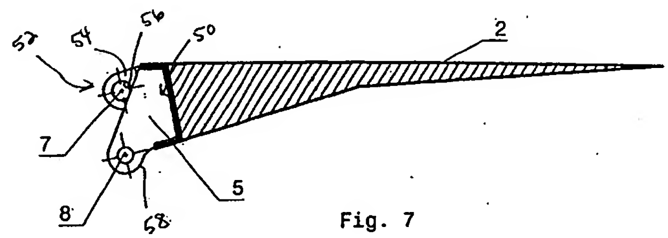


Fig. 7

The structural component connecting part of the fitting (5) includes at least one arm (34, 44, 54) extending outwardly in a direction away from the movable part mounting structure, with the arm having an aperture (36, 46, 56) therethrough (page 7, second, third and last paragraphs, as amended in the January 2009 Amendment). The inner diameter of the aperture defines a bearing surface (7) configured to receive at least one bearing (page 7, second, third and last paragraphs, as amended in the January 2009 Amendment). Glue is used to secure the movable part mounting structure of the fitting to the movable part (page 7, last paragraph, as amended in the January 2009 Amendment).

The fitting (5) is made of a synthetic composite material according to a resin transfer molding (RTM) method and

includes a carbon fabric as a reinforcement element (page 3, lines 27-29; page 4, lines 23-25 and lines 35-37; page 6, line 36 to page 7, line 13). The composite material is the same material as that from which the movable part is made (page 4, lines 7-9). The movable part is selected from the group consisting of a spoiler (2), a landing flap (3) and a control surface (4) (page 6, lines 12-15).

Appellant's invention as set forth in claim 27 is directed to a device for connecting a movable part (2, 3, 4) of an aircraft with a structural component (1) of the aircraft (page 1, lines 3-5). The device includes a fitting (5) having a movable part mounting structure (30, 40, 50) and a structural component connecting part (32, 42, 52) to connect the movable part to the structural component (page 7, second, third and last paragraphs, as amended by the January 2009 Amendment).

The structural component connecting part (32, 42, 52) of the fitting (5) includes at least one arm (34, 44, 54) extending outwardly in a direction away from the movable part mounting structure, with the arm having an aperture (36, 46, 56) therethrough (page 7, second, third and last paragraphs, as amended by the January 2009 Amendment). The inner diameter of the aperture defines a bearing surface (7) for receiving at least one bearing (page 7, second, third and last paragraphs, as amended by the January 2009 Amendment). Glue secures the movable

part mounting structure of the fitting to the movable part (page 7, last paragraph, as amended by the January 2009 Amendment). The movable part is selected from the group consisting of a spoiler (2), a landing gear (3) and a control surface (4) (page 6, lines 12-15). The fitting is made of the same composite material as the movable part (page 4, lines 7-9).

Appellant's invention as set forth in claim 30 is directed to the combination of a fitting (5) for connecting a movable part (2, 3, 4) of an aircraft with a structural component (1) of the aircraft, and the movable part itself (page 1 lines 3-5; page 6, lines 12-15).

The fitting (5) has a movable part mounting structure (30, 40, 50) and a structural component connecting part (32, 42, 52), with the structural component connecting part of the fitting including at least one arm (34, 44, 54) extending outwardly in a direction away from the movable part mounting structure and having an aperture (36, 46, 56) therethrough that defines a bearing surface (7) for receiving at least one bearing (page 7, second, third and last paragraphs as amended by the January 2009 Amendment). The fitting is made of a synthetic composite material according to the RTM method and includes a carbon fabric as a reinforcement element (page 3, lines 27-29; page 4, lines 23-25 and lines 35-37; and page 6, line 36 to page 7, line 13).

The movable part (2, 3, 4) is also made of the

composite material such that the fitting and the movable part have substantially the same thermal expansion coefficient (page 3, line 27 to page 4, line 2; page 4, lines 7-12). The movable part has an upper covering layer (16) and a lower covering layer (17) of fiber reinforced fabric, with the movable part mounting structure of the fitting being arranged therebetween and configured to connect the movable part to the structural component (page 8, lines 5-19). The movable part is selected from the group consisting of a spoiler (2), a landing flap (3) and a control surface (4) (page 6, lines 12-15).

VI. GROUNDS OF REJECTION TO BE REVIEWED ON APPEAL

Rejections under 35 U.S.C. §103(a)

Grounds of rejection to be reviewed on appeal include 1) whether independent claims 15, 27 and 30 are unpatentable under 35 U.S.C. §103(a) over U.S. Patent No. 6,234,423 to Hirahara et al. ("Hirahara") in view of U.S. Patent No. 5,224,670 to Padden; and 2) whether the process by which the product set forth in independent claims 15 and 30 is produced bears on patentability.

The patentability of the dependent claims is not being separately argued herein with the understanding that, should the Board find independent claims 15, 27 and 30 to be patentable over Hirahara and Padden, the patentability of the dependent claims will follow due to their dependent relationship from the independent claims.

VII. ARGUMENT

The final rejections of claims 15, 27 and 30 in the present application are based on 35 U.S.C. §103(a). Under U.S. patent law, the burden is on the Examiner to establish a prima facie case of obviousness of the claimed subject matter over prior art references. In re Deuel, 51 F.3d 1552, 1557, 34 USPQ2d 1210, 1214 (Fed. Cir. 1995). Only after that burden is met must the applicant come forward with arguments or evidence in rebuttal. Id. To establish prima facie obviousness of a claimed invention, all the claim limitations must be taught or suggested by the prior art. In re Royka, 490 F.2d 981, 180 USPQ 580 (CCPA 1974).

A. The Combination of Hirahara and Padden Cannot Render Appellant's Claims 15, 27 and 30 Prima Facie Obvious Under 35 U.S.C. §103(a) Because the Combination of References Does Not Teach Each and Every Element of Appellant's Claimed Invention

As set forth in claim 15, the present invention is directed to a connecting device used in an aircraft to connect a structural component of the aircraft with a movable part. The movable part is defined as being a spoiler, a landing flap or a control surface, each of which places extremely high loads upon

the connecting device. The connecting device includes at least one fitting that is made of a synthetic composite material according to a resin transfer molding (RTM) method and includes a carbon fabric as a reinforcement element.

The composite material is the same material from which the movable part is made. With this construction, the present invention reduces the demand on the connection between the fitting and the movable part in that both elements, being made of the same composite material, share substantially the same thermal expansion coefficient. In addition, the fitting is secured to the movable part by gluing.

Claim 27 provides a device for connecting a movable part of an aircraft with an aircraft structural component. Like claim 15, the movable part is a spoiler, a landing flap or a control surface. The device includes a fitting made of the same composite material as the movable part. The fitting has a movable part mounting structure that is secured to the movable part using glue, and a structural component connecting part that is secured to the structural component.

To provide articulation, the structural component connecting part of the fitting includes at least one arm having an aperture therethrough that extends outwardly in a direction away from the movable part mounting structure. The inner diameter of the aperture defines a bearing surface for receiving

at least one bearing.

Claim 30 is directed to the combination of a fitting and a movable part. The fitting is configured to connect a movable part of an aircraft with a structural component of the aircraft. The fitting has a movable part mounting structure and a structural component connecting part, the latter of which includes at least one arm extending outwardly in a direction away from the movable part mounting structure and having an aperture therethrough that defines a bearing surface for receiving at least one bearing. The fitting is made of a synthetic composite material according to the RTM method and includes a carbon fabric as a reinforcement element.

The movable part in claim 30 is further defined as being made of the same composite material as the fitting such that the fitting and the movable part have substantially the same thermal expansion coefficient. The movable part includes an upper covering layer and a lower covering layer of fiber reinforced fabric, and the movable part mounting structure of the fitting is arranged between the upper and lower covering layers and is configured to connect the movable part to the structural component. As in claims 15 and 27, the movable part is a spoiler, a landing flap or a control surface.

With the foregoing structure, the fitting according to the present invention provides a structure suitable for

connecting movable parts subject to high shear stress with structural components of airplanes. By making the fitting of composite material instead of conventional metal, the difficulties associated with the thermal expansion of known metal devices are avoided. In addition, the composite material fitting offers a low weight, a high loading capacity, and a simplified producibility. Such a fitting is not shown by the prior art.

1. <u>Hirahara Does Not Show the Use of Composite Material for a Fitting as Provided in Claims 15, 27 and 30</u>

As set forth in greater detail above, claims 15, 27 and 30 are directed to a fitting for connecting a movable part of an aircraft with a structural component of the aircraft. Accordingly, there are three components, namely the movable part, the structural component and the fitting. The fitting acts as a hinge between the other two parts and includes a movable part mounting structure for connection to the movable part and a structural component connecting part for connection to the structural component.

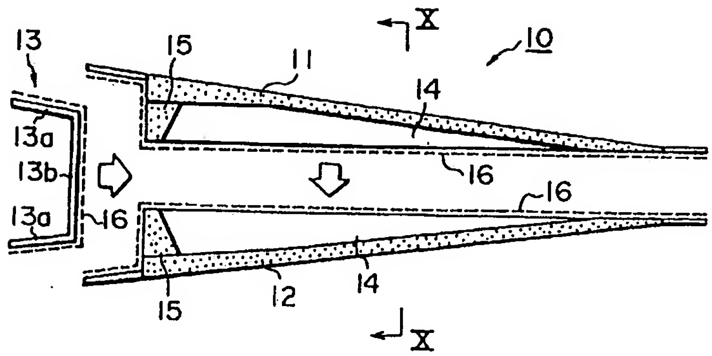
To provide the articulating or hinged connection between the structural component connecting part of the fitting and the structural component of the aircraft, the structural component connecting part of the fitting includes at least one

arm that extends outwardly in a direction away from the moving part mounting structure. The arm has an aperture therethrough that forms the bearing surface for at least one bearing which supports the movement of the moving part. The bearing itself may be a sphere or roller such as a ball bearing, or a cylinder such as a sleeve bearing.

As defined in each of claims 15, 27 and 30, the structural component connecting part as claimed is part of the fitting and the arm with the aperture therein that defines the bearing surface is part of the structural component connecting part. Since the claims provide that *the fitting* is made of a composite material, it follows by necessity that the structural component connecting part with its arm, aperture and bearing surface is made of a composite material. Such a fitting is not shown by Hirahara.

Hirahara is directed to a method of making an airfoil structure 10 of composite prepreg material. For ease of reference, Figure 9 of Hirahara is reproduced on the next page.

FIG. 9



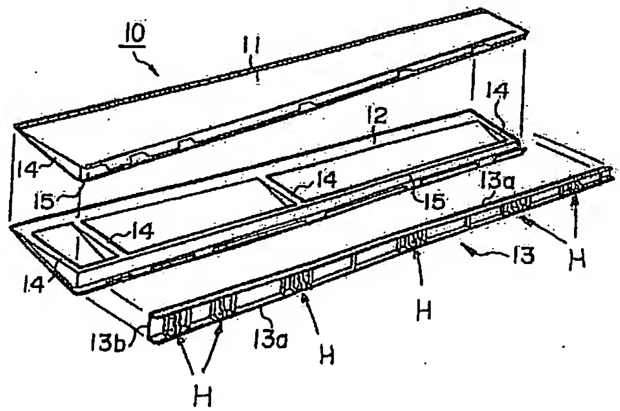
The composite airfoil structure of Hirahara includes a U-shaped spar 13 bonded to upper and lower skins 11 and 12 (see column 4, lines 50-60). The spar 13 includes a web 13b with flanges 13a on each side of the web that together form the U-shaped cross section of the spar (see Hirahara, column 5, lines 23-25; and Figures 9 and 14).

Neither 13a nor 13b are used *to connect* the airfoil structure to the aircraft structural component; rather, elements 13a and 13b are part of the spar which, in turn, is part of the airfoil structure itself (see column 4, lines 57-59). Hence, the airfoil structure 10 corresponds with the movable part of Appellant's invention.

The actual connecting structure or fitting used to connect the airfoil 10 to the aircraft wing is not identified by name in Hirahara but is simply illustrated in Figures 1 and 2

thereof as a hinge or arm projecting outwardly from the web 13b and between the flanges 13a. Figure 2 of Hirahara is reproduced below and has been annotated to identify the hinges or arms with the letter "H".

FIG. 2



The hinge or arm "H", while mounted to the spar, is a separate component from the spar and it is this unnamed arm which corresponds with the claimed fitting of the Appellant's invention, and not the spar 13.

Contrariwise, in rejecting the claims, the Examiner concluded that the spar 13 is a fitting, stating that the fitting corresponds with the components identified in Hirahara by reference numerals 13a and 13b. To reach this conclusion, the Examiner has taken the position that the web 13b of the spar 13 includes the unnamed arm or hinge "H". Since the spar web 13b and flanges 13a are disclosed in Hirahara as being made of a composite material, the Examiner goes on to conclude that

Hirahara also discloses that the arm or hinge "H" is also made of composite material when, in fact, this is not shown or suggested by Hirahara.

As just explained and as explicitly taught by Hirahara, the spar is a *U-shaped* structure comprised of the web and two flanges (see Figures 9 and 14 of Hirahara). The very method of producing the spar taught in Hirahara makes it clear that the arm is not a part of the spar. Specifically, in forming the spar, a laminate of composite prepreg is formed into the illustrated U-shaped cross sectional shape and molded under heat (see column 6, lines 6-11). The spar is then fitted to the upper and lower skins, as shown in Figure 9, and pressed by load application blocks 34 in a bonding jig 30 to form a *single airfoil structure* (see Figures 11 and 12; column 6, lines 18-29). The pressing of the load application blocks 34 against the spar could not occur if, as concluded by the Examiner, the spar included an arm extending outwardly from the web and integral therewith.

Beyond the bare depiction of the arm or fitting "H" in the structure illustrated in Figures 1 and 2, Hirahara does not address the fittings at all. This was evidenced by the Appellant during prosecution by the Declaration of Helmut Kaufmann ("the Kaufmann Declaration"), which is attached in the Evidence Appendix.

In paragraph 8 of the Kaufmann Declaration, Mr.

Kaufmann refers to the fitting as a hinge and, in fact, it was Mr. Kaufmann who annotated Figure 2 of Hirahara in the manner shown above. Specifically, Mr. Kaufmann in the Kaufmann Declaration identified the fitting/hinge in Hirahara as being represented by the six unlabeled arms of Hirahara, Figure 2, and he marked these hinges as shown in Figure 2 with the letter "H" (see the attachment to the Kaufmann Declaration). In discussing these fittings/hinges in paragraph 8 of the Kaufmann Declaration, Mr. Kaufmann states that to his understanding, "*Hirahara does not address the hinges at all*, but only methods of forming the box structure of the movable surface" (emphasis added). Therefore, the Kaufmann Declaration substantiates that elements 13a and 13b are part of the box structure of the movable part in Hirahara; they are not fittings as that term is used by Appellant and known and understood in the aircraft industry. The Examiner has offered no evidence to rebut the Kaufmann Declaration.

Since the spar in Hirahara does not include the fittings or hinges, but only components 13a and 13b, then it is clear that Hirahara only discloses an airfoil structure, or movable part, made of composite material. There is nothing at all in Hirahara to suggest that the unnamed hinges in Figures 1 and 2 of Hirahara, which are used to connect the spar to the aircraft structure, are made of a composite material. Nor would the composite prepreg material used to make the spar as taught by

Hirahara have sufficient strength to be used as a fitting (see Kaufmann Declaration, paragraph 12). Hence, the skilled person would have no reason to conclude that the fittings in Hirahara are anything other than conventional metal connectors.

Therefore, Hirahara does not disclose a fitting made of a composite material as set forth in each of claims 15, 27 and 30. Hirahara also does not disclose a fitting made of the same composite material as the movable part as provided in claim 27.

2. <u>Padden Does Not Show the Use of Composite Material for a Fitting as Provided in Claims 15, 27 and 30</u>

Padden is directed to a spoiler, or movable part of an aircraft, made of graphite epoxy. As such, the scope of Padden's disclosure is analogous to that of Hirahara, being limited to the description of an *airfoil structure*. Padden more clearly differentiates the airfoil structure from the fittings, however, and is consistent with Appellant's description of what is shown in Hirahara. Figure 1 of Padden is reproduced above.

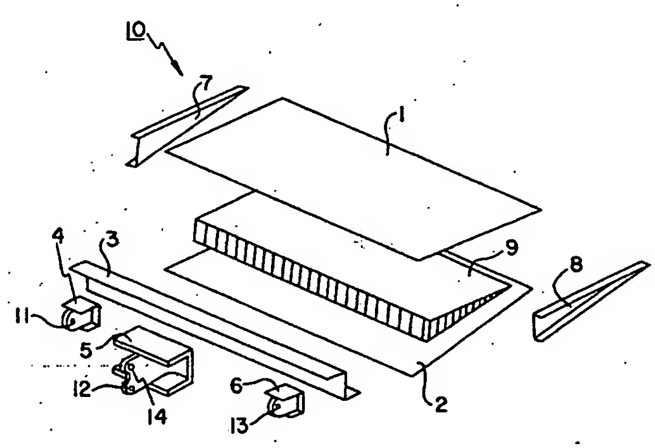


FIG.1

As described in Padden, column 1, line 62, to column 2, line 7, with reference to Figure 1 as above, the spoiler has six parts, namely the upper cover 1, the lower cover 2, the spar 3, two pre-cured closure ribs 7 and 8, and the core 9. The fittings 4, 5 and 6, which *are* specifically identified with reference numerals in Padden, unlike in Hirahara, are clearly NOT part of the spoiler. Rather, just as in Hirahara, the fittings in Padden are separate components that are attached to the spar (column 2, line 4 of Padden) and are used to connect the spoiler to the aircraft wing. And it is the fittings that define the hinge apertures 11-13 that provide the articulating joint or hinge which allows the spoiler to move with respect to the wing or structural component of the aircraft when connected thereto by the fittings.

Finally, as affirmatively stated in Padden, the mounting fittings 4, 5, 6 and 25, 26 and 27 are made of metal, preferably aluminum (see column 4, lines 9-11). Thus, there is nothing in Padden to suggest a fitting made of composite material as set forth in claims 15, 27 and 30 of the present invention. Nor is there anything in Padden to suggest making a fitting of the same composite material as the movable part as provided in claim 27.

3. It Would Not Have Been Obvious to Use Composite Synthetic Material Formed by the RTM Method for a Fitting as Provided in Claims 15 and 30

As substantiated by the Declaration of Wolfgang Billinger ("the Billinger Declaration"), which is attached in the Evidence Appendix, in this field of technology it was neither well known nor obvious to use a composite material for the fitting of a spoiler, landing flap or control surface.

On the contrary, as stated in paragraph 6 of the Billinger Declaration, persons in the aviation field recognized that a fitting suitable for high load application such as that encountered during take off and landing, must be particularly stable to resist shearing forces. To achieve such stability, the only known practice was to rely on metal fittings secured with metal fasteners (see paragraph 11 of the Billinger Declaration).

Further, while it was known to use synthetic composite material for structural components such as the wing, composite parts made according to the RTM method were considered structurally inadequate for fittings (see paragraph 8 of the Billinger Declaration).

That persons of ordinary skill in the art considered hinges or fittings made of carbon fiber reinforced plastic according to the RTM method to be wholly unsuitable for an aircraft was further substantiated in the Declaration of Helmut

Kaufmann & Rudolf Gradinger ("the Kaufmann/Gradinger Declaration"), which is also attached in the Evidence Appendix.

More particularly, Mssrs. Kaufmann and Gradinger state in paragraphs 6-8 of the Kaufmann/Gradinger Declaration that, prior to their learning of Appellants' invention in 2004, it was conventional to make aircraft spoiler hinges of metal, and to use metal fasteners to connect the metal fittings securely to the movable parts. Mssrs. Kaufmann and Gradinger state in the Kaufmann/Gradinger Declaration that they were surprised when they heard of the design for a carbon fiber reinforced plastic/resin transfer molding aircraft spoiler hinge, as claimed by the present invention, further stating that an RTM approach for a hinge or fitting designed for use with aircraft spoilers had not previously been used.

Clearly, and as evidenced by the Kaufmann/Gradinger Declaration, while the RTM method was known for other applications, persons of ordinary skill in the art did not consider a hinge fitting made of synthetic composite material according to the RTM method to be suitable for the intended use of a high-load aircraft fitting. Therefore, selection of such material and method was not a matter of obvious design choice but instead was a non-obvious departure from known solutions. The Examiner has offered no evidence to rebut either the Kaufmann/Gradinger Declaration or the Billinger Declaration in

regard to these facts.

Therefore, neither Hirahara nor Padden would suggest to the skilled person that a fitting could be made of a synthetic composite material by the RTM method as set forth in claims 15 and 30.

4. <u>Neither Hirahara nor Padden Disclose the Use of Glue to Secure the Fitting to the Movable Part as Provided in Claims 15 and 27 and It Would Not Have Been Obvious to Use Glue to Hold the Fitting</u>

As substantiated by the Billinger Declaration, in this field of technology it was neither well known nor obvious to secure a fitting to the movable part of an aircraft by gluing.

As disclosed in Hirahara, it was known to use glue when adhering composite prepreg laminate layers to one another to form an airfoil structure (see column 4, line 60 to column 5, line 6). However, Hirahara is silent with respect to how the fitting is secured to the spar. But in any event, the use to which glue is put in Hirahara is not comparable to the use of glue for securing a fitting to a movable part.

On the contrary, just as composite materials were not considered by persons skilled in the art to provide adequate strength for use in forming a fitting, the use of glue to secure a fitting to the movable parts of an aircraft was also considered

to be inadequate to withstand the known shearing forces to which a fitting is subjected (see paragraphs 8 and 11 of the Billinger Declaration). "Instead, what was known by persons of ordinary skill was reliance on metal fasteners, often a high number of screws or rivets, for secure connection of metal fittings to movable parts." (Billinger Declaration, paragraph 6).

The foregoing fact is consistent with what is taught in Padden. Particularly, it can be surmised from Figure 2 of Padden that the metal fittings disclosed therein are connected to the movable part or spoiler with metal connecting elements that extend through the fitting, such as rivets or the like. There is nothing to suggest that glue is used to connect the fitting in Padden.

In addition, because only metal fasteners were considered to have sufficient strength to secure a fitting, gluing was not considered to be a fastening technique equivalent to metal fasteners for high-load aircraft applications (see paragraph 12 of the Billinger Declaration). The Examiner has offered no evidence to rebut the Billinger Declaration in regard to this further fact.

Mssrs. Kaufmann and Gradinger also state in the Kaufmann/Gradinger Declaration that the fact that glue was found to have sufficient strength to withstand the known shearing forces to which movable aircraft parts are subject was unexpected

(see paragraph 11). Had the use of glue been an obvious alternative to metal fasteners, the skilled person would certainly have used it previously since, as also stated in paragraph 11 of the Kaufmann/Gradinger Declaration, the use of glue rather than conventional screws or rivets is beneficial as it reduces the weight of the aircraft which is an important consideration in aviation.

In sum, neither Hirahara nor Padden disclose the use of glue to secure the fitting and it would not have been obvious to the skilled person that glue could be used to effectively secure a fitting as set forth in claims 15 and 27.

<u>B. The Process by Which the Product Set Forth in Appellant's Claims 15 and 30 is Produced Does Bear on Patentability Because the Resulting Product According to the Claimed Invention Has Distinctive Structural Characteristics</u>

As set forth in Section 2113 of the U.S. Manual of Patent Examining Procedure (MPEP), a product-by-process claim is a structure claim and therefore the product must distinguish over the prior art on the basis of its structure. However, when the manufacturing steps used to make the product would be expected to impart distinctive structural characteristics to such product, the structure implied by the process steps should be considered when assessing the patentability of a product-by-process claim.

See, e.g., *In re Garnero*, 412 F.2d 276, 279, 162 U.S.P.Q. 221, 223 (CCPA 1979).

Independent claims 15 and 30 each specify that the fitting is made by the RTM method. A fitting made by this process has distinctive structural characteristics that are distinguishable over fittings made by other processes as discussed in the specification on page 4, lines 23-34; and in the last paragraph on page 6 which extends onto page 7.

As set forth in paragraph 11 of the Kaufmann Declaration, the resins used in fiber reinforced systems such as RTM are modified with additives that change the structure of the resin and produce a part having mechanical properties similar to forged versions made from alloys of aluminum, high strength steel or titanium. These enhanced mechanical properties are present in the claimed fittings and are needed "when the fittings will be used in parts which have significant wall thickness differentiation, as is the case with aircraft fittings and associated movable parts" (see paragraph 10).

In addition, the RTM method provides particular benefit in that it offers more degrees of freedom in aligning the fibers along the main stresses. Thus, according to the Kaufmann/Gradinger Declaration, paragraph 9, the RTM method is the first composite manufacturing method appropriate for bulky structurally loaded parts. Hence, the fitting made by the RTM

method is clearly distinguishable structurally over composite parts made by other methods such as the airfoil structures shown in Hirahara and Padden.

Hirahara discloses an airfoil formed by laminating various layers of composite prepreg on a forming jig and then hardening the assembly by applying heat (see column 4, line 60 to column 5, line 6). As already explained, Hirahara does not address the hinges/fittings and, contrary to the Examiner's view, does not disclose that the fittings are made of a composite material. However, as stated in the Kaufmann Declaration, "if one were to make the hinges of Hirahara of the same composite material as that disclosed for the skins and spar 13, such theoretical hinges would lack sufficient structural strength for their intended use" (see paragraph 12). Again, the fittings made by the RTM method are structurally different from composite parts made by other methods.

In paragraphs 9 and 10 of the Kaufmann Declaration, Mr. Kaufmann states that, due to the significant wall thickness differences between hinges and airfoil web and flange structures, the spar 13 of Hirahara could not be produced as a composite part together with the fitting using the composite prepreg structure taught by Hirahara. Only the RTM method can produce a hinge fitting having the required mechanical properties to be used in place of a metal fitting to connect the movable part to the

structural component of the aircraft. Therefore, the fitting made by the RTM method as set forth in claims 15 and 30 has enhanced mechanical properties over, and "is structurally different from", the composite prepreg structure taught in Hirahara for the airfoil (see paragraph 12 of the Kaufmann Declaration). The Examiner has offered no evidence to rebut this further fact from the Kaufmann Declaration.

Padden discloses a composite movable part connected to the wing using metal fittings. Therefore, Padden contributes nothing that would suggest an aircraft fitting made of composite material by the RTM method.

VIII. CONCLUSION

In conclusion, Appellant respectfully requests reversal of the Examiner's rejection of claims 15, 27 and 30 under 35 U.S.C. §103(a) as being unpatentable over Hirahara in view of Padden for the reasons discussed above.

Respectfully submitted,

JACOBSON HOLMAN PLLC

Date: April 7, 2010 By: ______________________

Customer No. 000,136
400 Seventh Street, N.W.
Washington, D.C. 20004
(202) 638-6666

Harvey B. Jacobson, Jr.
Registration No. 20,851

Suzin C. Bailey
Registration No. 40,495

IX. CLAIMS APPENDIX

All claims involved in this appeal appear below.

The pending claims are as follows:

Claims 1-14 (Canceled).

15. (Previously Presented) A connecting device used in an aircraft to connect a movable part of said aircraft with a structural component of said aircraft, said connecting device comprising at least one fitting having a movable part mounting structure and a structural component connecting part so as to be configured to connect said movable part with said structural component, said structural component connecting part of said fitting including at least one arm extending outwardly in a direction away from said movable part mounting structure and having an aperture therethrough, an inner diameter of said aperture defining a bearing surface configured to receive at least one bearing, and glue securing said movable part mounting structure of said fitting to said movable part, said fitting being made of a synthetic composite material according to a resin transfer molding method and including a carbon fabric as a reinforcement element, said composite material being a same

material as that from which said movable part is made, said movable part being selected from the group consisting of a spoiler, a landing flap and a control surface.

Claims 16-18. (Canceled).

19. (Previously Presented) The device as set forth in claim 15, further comprising a reactive material sewn or woven in said carbon fabric.

20. (Previously Presented) The device as set forth in claim 19, wherein said reactive material is nylon.

21. (Previously Presented) The device as set forth in claim 15, further comprising a recess provided in said movable part, said movable part mounting structure of said fitting being embedded in said recess.

22. (Previously Presented) The device as set forth in claim 21, wherein said movable part includes an upper covering layer and a lower covering layer, said movable part mounting structure of said fitting being arranged therebetween.

23. (Previously Presented) The device as set forth in claim

22, wherein said movable part mounting structure of said fitting is glued with said upper covering layer and said lower covering layer.

24. (Withdrawn) The device as set forth in claim 15, further comprising additional connecting means.

25. (Withdrawn) The device as set forth in claim 24, wherein said additional connecting means include rivets or screws.

26. (Previously Presented) The device as set forth in claim 15, further comprising an indentation provided in said movable part, wherein said movable part mounting structure of said fitting is embedded in said indentation.

27. (Previously Presented) A device for connecting a movable part of an aircraft with a structural component of said aircraft, said device comprising a fitting having a movable part mounting structure and a structural component connecting part to connect said movable part to said structural component, said structural component connecting part of said fitting including at least one arm extending outwardly in a direction away from said movable part mounting structure and having an aperture

therethrough, an inner diameter of said aperture defining a bearing surface for receiving at least one bearing, and glue that secures said movable part mounting structure of said fitting to said movable part, said movable part being selected from the group consisting of a spoiler, a landing gear and a control surface, and said fitting being made of a same composite material as said movable part.

28. (Withdrawn) A device for connecting a movable part of an aircraft with a structural component of said aircraft, said device comprising a fitting integrally formed as one piece with said movable part and made of a same synthetic material as said movable part, said fitting having a structural component connecting part configured to connect said integral fitting and movable part with said structural component, said structural component connecting part including at least one arm extending outwardly in a direction away from said movable part and having an aperture therethrough defining a bearing surface for connection with said structural component, said synthetic material made according to a resin transfer molding method and including a reinforcing carbon fabric, said movable part being selected from the group consisting of a spoiler, a landing flap and a control surface.

29. (Canceled).

30. (Previously Presented) The combination of a fitting for connecting a movable part of an aircraft with a structural component of said aircraft, and said movable part, comprising:

said fitting having a movable part mounting structure and a structural component connecting part, said structural component connecting part of said fitting including at least one arm extending outwardly in a direction away from said movable part mounting structure and having an aperture therethrough that defines a bearing surface for receiving at least one bearing, said fitting being made of a synthetic composite material according to a resin transfer molding method and including a carbon fabric as a reinforcement element;

said movable part also being made of said composite material such that said fitting and said movable part have substantially a same thermal expansion coefficient, said movable part having an upper covering layer and a lower covering layer of fiber reinforced fabric, and said movable part mounting structure of said fitting being arranged therebetween and configured to connect said movable part to said structural component, said movable part being selected from the group consisting of a spoiler, a landing flap and a control surface.

31. (Canceled).

32. (Previously Presented) The combination as set forth in claim 30, further comprising glue applied to said fitting and/or said movable part to secure said fitting to said movable part.

33. (Previously Presented) The device as set forth in claim 27, wherein said glue alone secures said fitting to said movable part, there being an absence of any other fastening component.

34. (Previously Presented) The device as set forth in claim 30, wherein said fitting and said movable part are integrally formed as one piece.

35. (Previously Presented) The device as set forth in claim 15, wherein said fitting further includes a second arm having an aperture therethrough that defines an articulation point.

36. (Previously Presented) The device as set forth in claim 35, wherein said arm defining the bearing surface is substantially parallel with said second arm defining the articulation point.

X. EVIDENCE APPENDIX

See attached references Hirahara and Padden, and attached Declaration of Wolfgang Billinger, Declaration of Helmut Kaufmann and Rudolf Gradinger, and Declaration of Helmut Kaufmann.

The Declaration of Wolfgang Billinger was filed concurrently with an Amendment and entered on the record by the Examiner on July 9, 2004.

The Declaration of Helmut Kaufmann and Rudolf Gradinger was filed concurrently with an Amendment and entered on the record by the Examiner on May 6, 2005.

The Declaration of Helmut Kaufmann was filed concurrently with an Amendment and entered on the record by the Examiner on August 2, 2007.

US006234423B1

(12) United States Patent
Hirahara et al.

(10) Patent No.: US 6,234,423 B1

(45) Date of Patent: May 22, 2001

(54) COMPOSITE AIRFOIL STRUCTURES AND THEIR FORMING METHODS

(75) Inventors: Makoto Hirahara; Yasuo Isano; Ryuhei Shimizu; Kazuaki Amaoka, all of Tokyo-To (JP)

(73) Assignee: Japan Aircraft Development Corporation, Tokyo-To (JP)

(*) Notice: Subject to any disclaimer, the term of this patent is extended or adjusted under 35 U.S.C. 154(b) by 0 days.

(21) Appl. No.: 09/363,396

(22) Filed: Jul. 29, 1999

(30) Foreign Application Priority Data

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(51) Int. Cl.[7] B64C 3/20

(52) U.S. Cl. 244/123; 416/226; 416/230; 244/133

(58) Field of Search .... 244/117 R, 123, 244/133, 131; 416/230, 226, 241 A; 156/245, 182; 29/889.71, 889.61, 889.6

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Primary Examiner—Galen L. Barefoot
(74) *Attorney, Agent, or Firm*—Smith, Gambrell & Russell, LLP

(57) ABSTRACT

A box-structure airfoil is constructed of a composite material upper skin, a composite material lower skin **12** and a spar. Ribs and an elongate projection are formed integrally on the inner surface of each of the two skins. The upper and lower skins and the spar are simultaneously bonded by an adhesive to form a single structure. Since the ribs and the elongate projection are one-piece formed with each skin, it is possible to reduce the number of principal constituent components. Furthermore, since assembly operation is made by using the adhesive, there is no need for fasteners or the like for assembling. Moreover, the spar is bonded not only to Moreover, because not only a web but also flanges on both sides of the spar are bonded to the respective skins, it is possible to obtain a large strength. According to the invention, it is possible to reduce the number of principal constituent components and that of assembling components of an airfoil, thereby achieving its cost reduction.

7 Claims, 13 Drawing Sheets

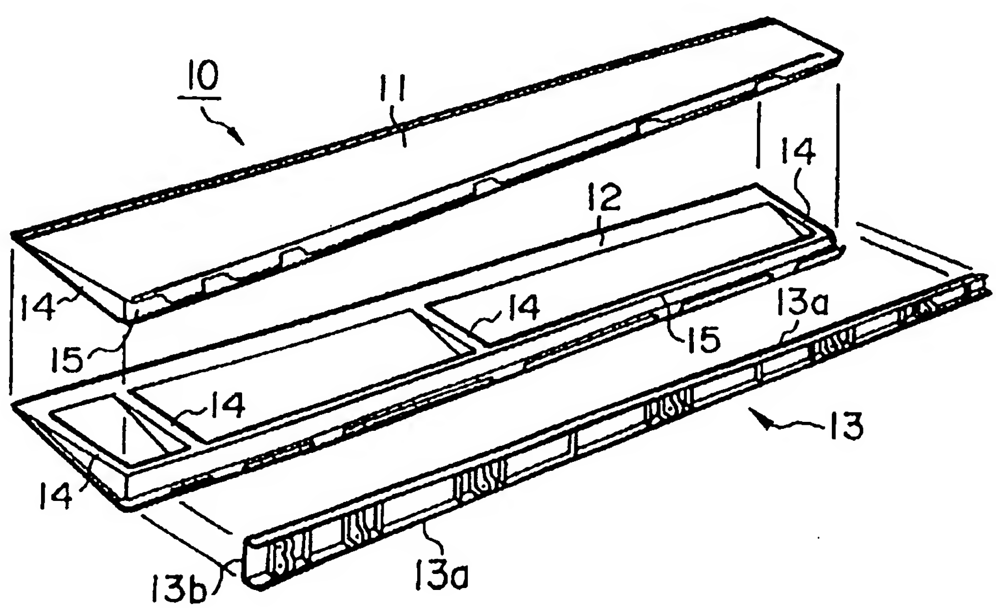
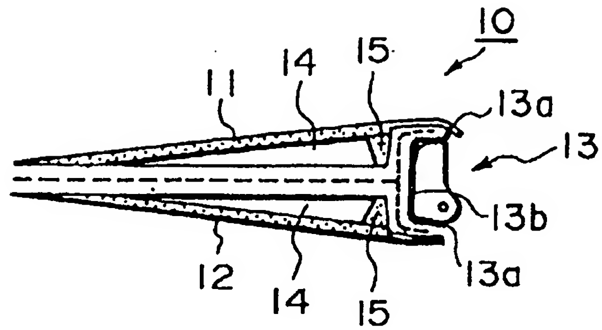


FIG. 1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13a
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13a
12
14
15

FIG. 2

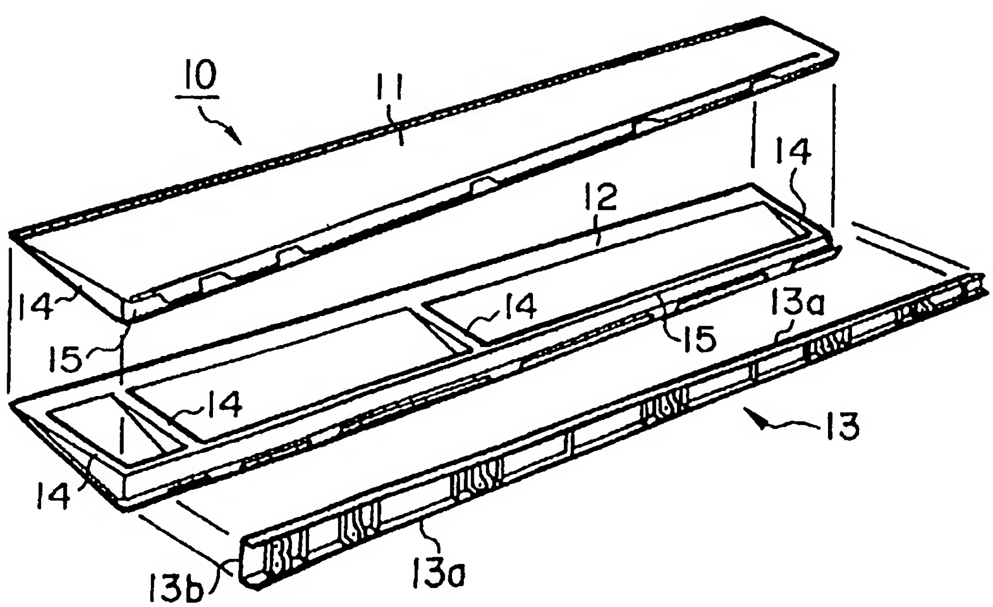
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FIG. 3

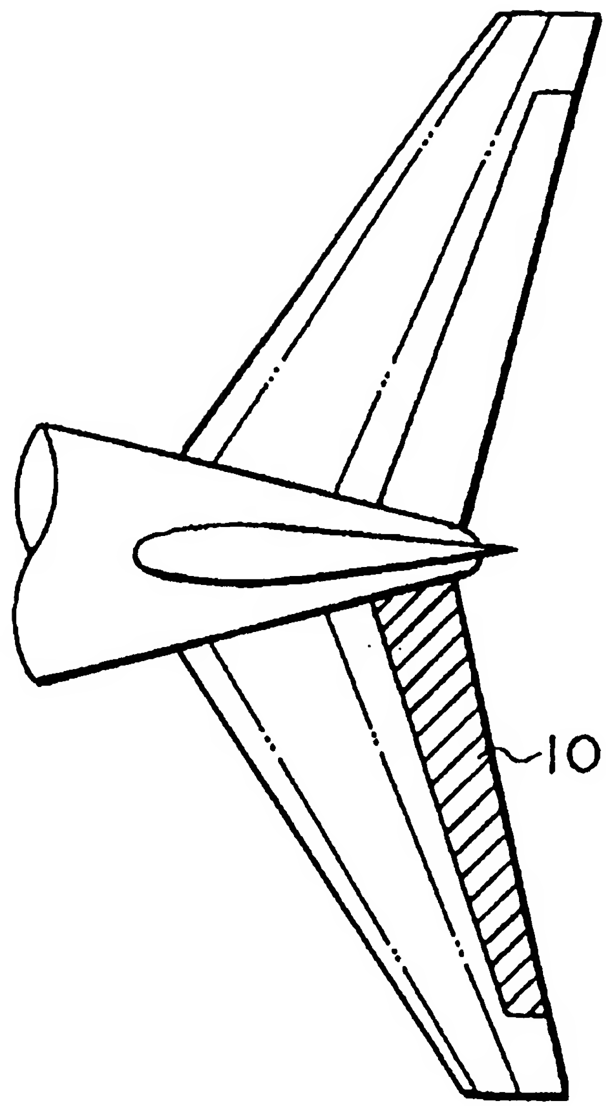


FIG. 4

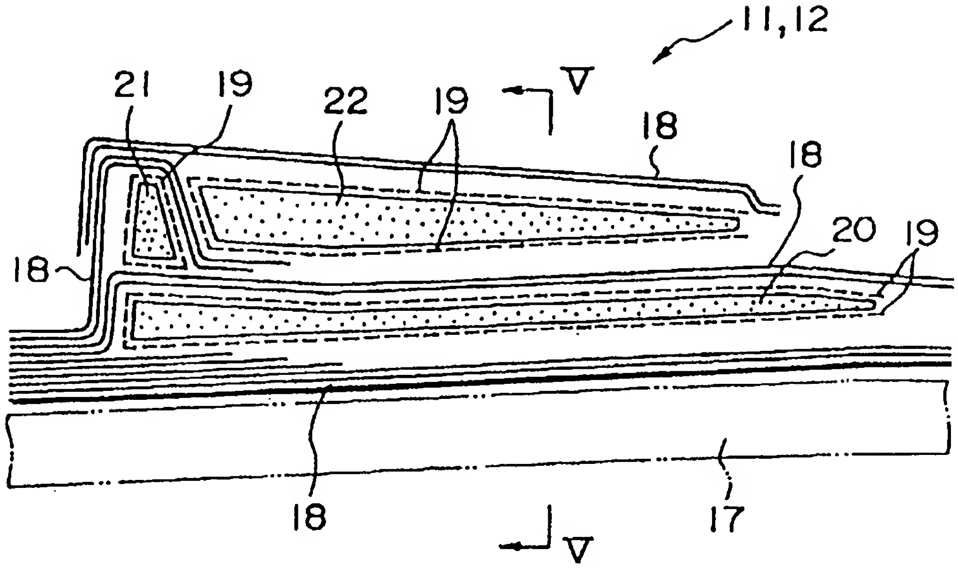


FIG. 5

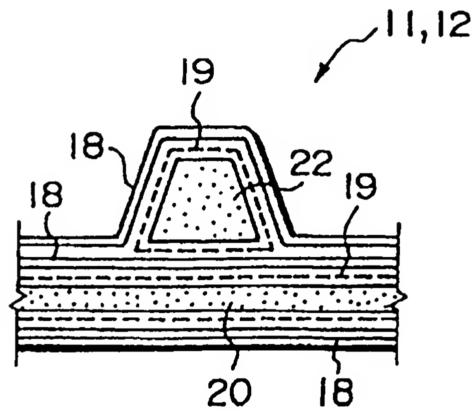


FIG. 6

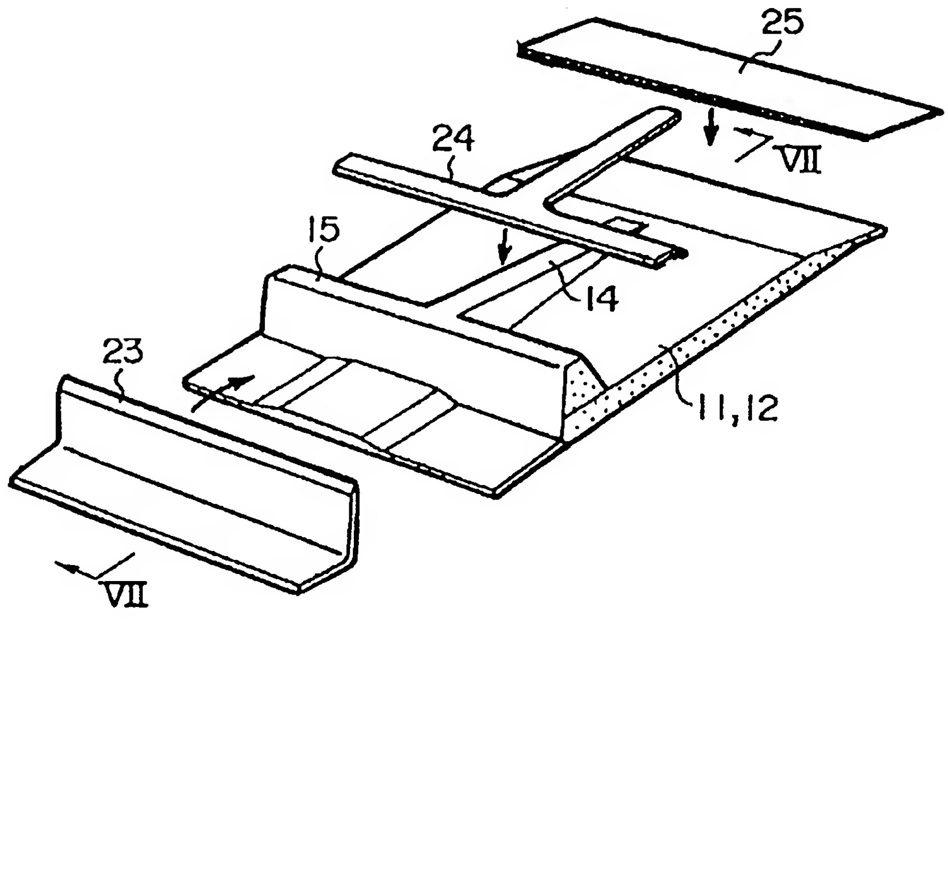


FIG. 7

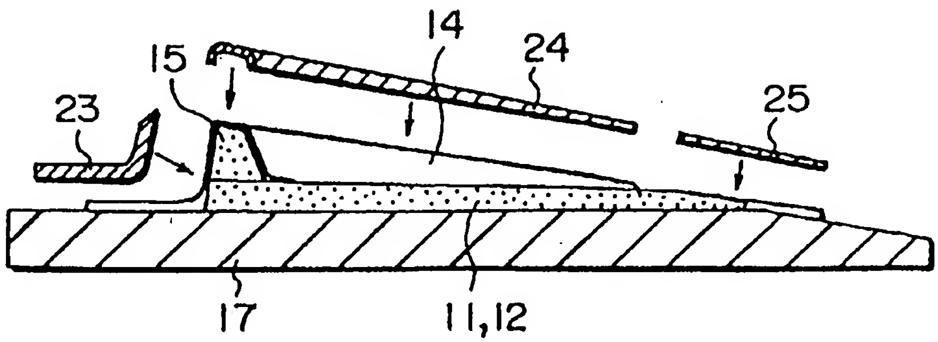


FIG. 8

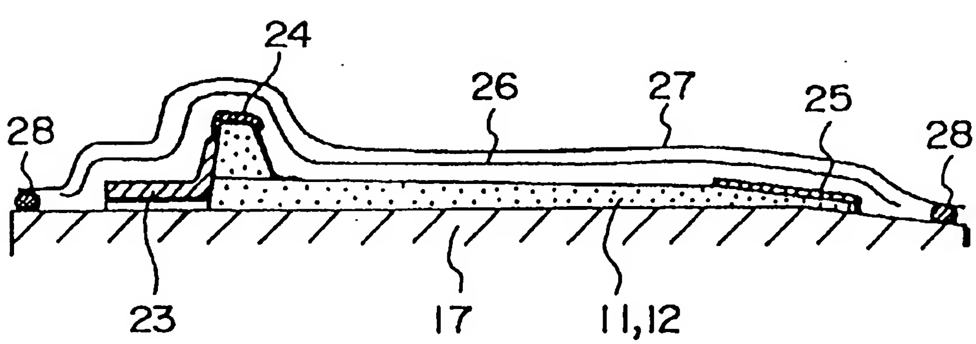


FIG. 9

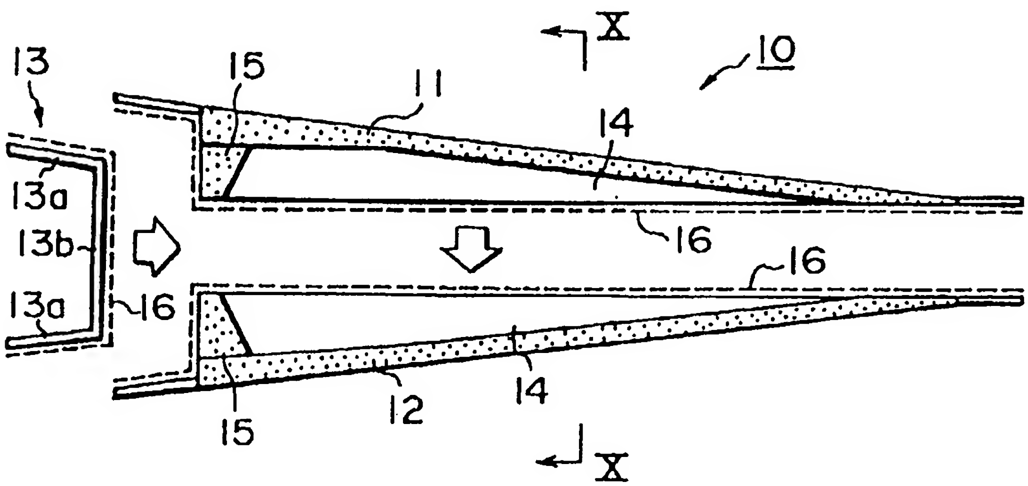


FIG. 10

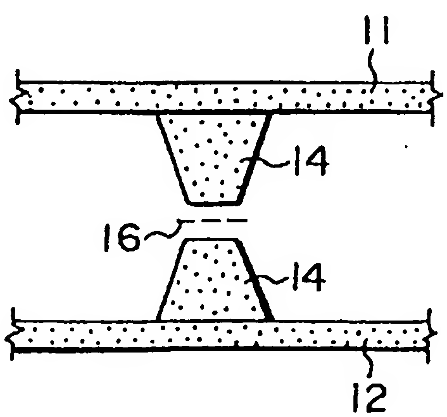


FIG. 11

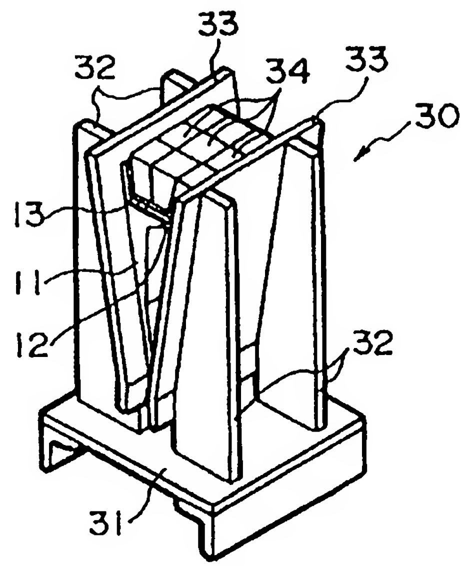


FIG. 12

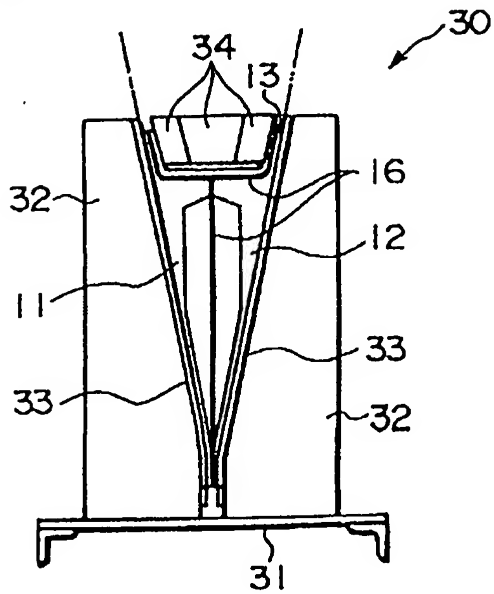


FIG. 13 (a)

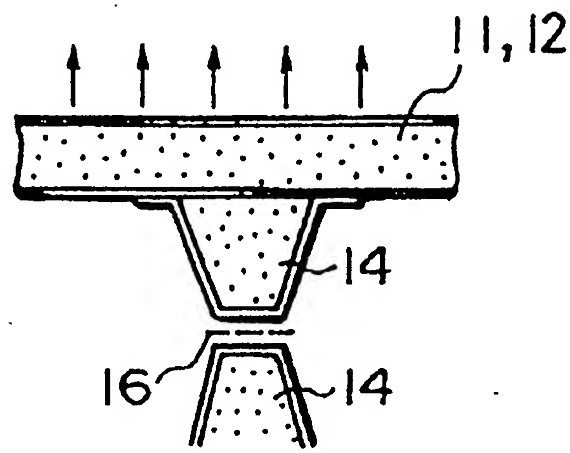


FIG. 13 (b)

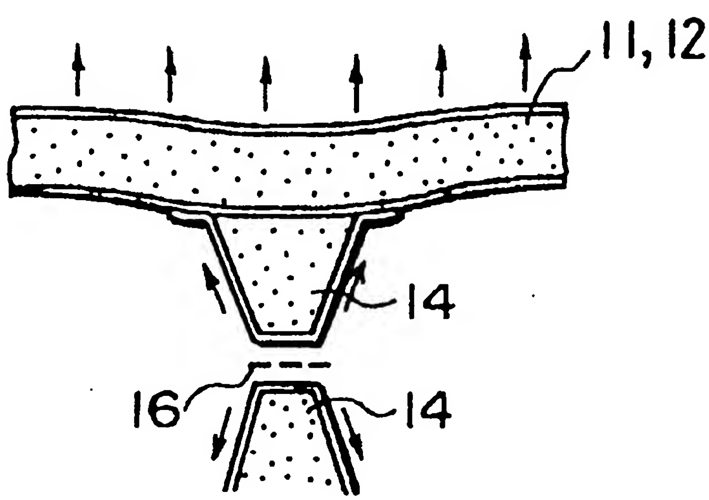


FIG. 14

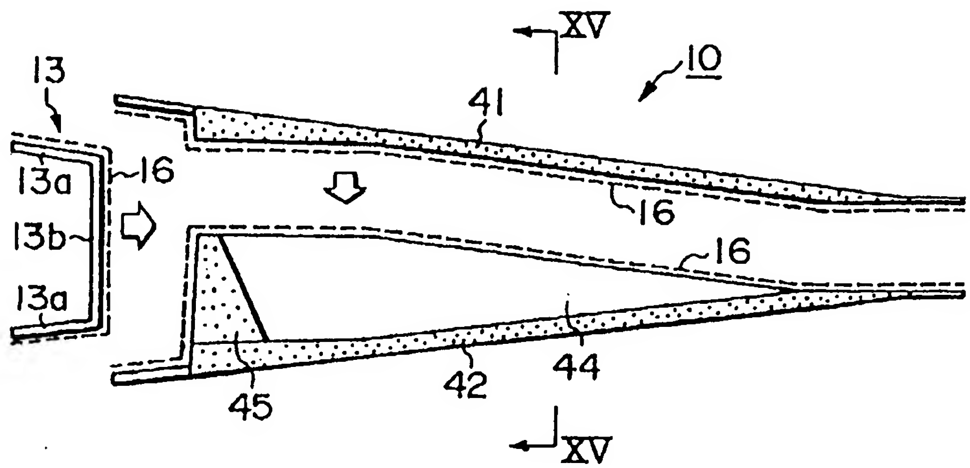


FIG. 15

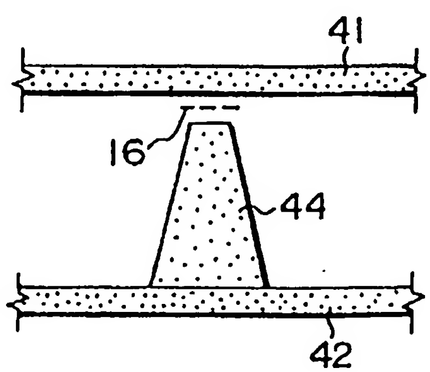


FIG. 16

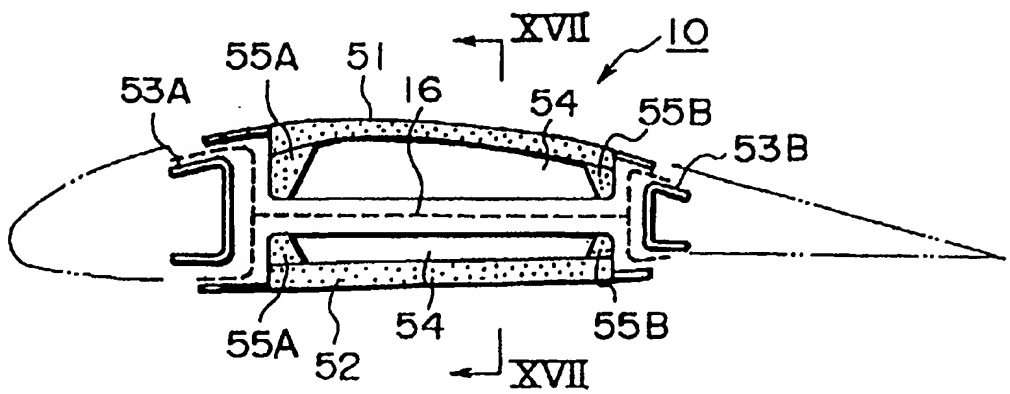


FIG. 17

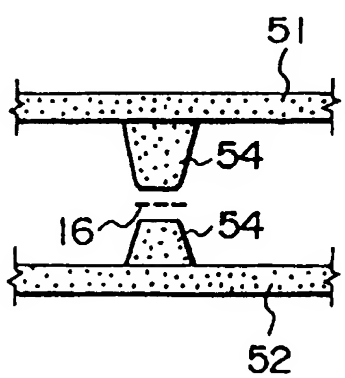


FIG. 18

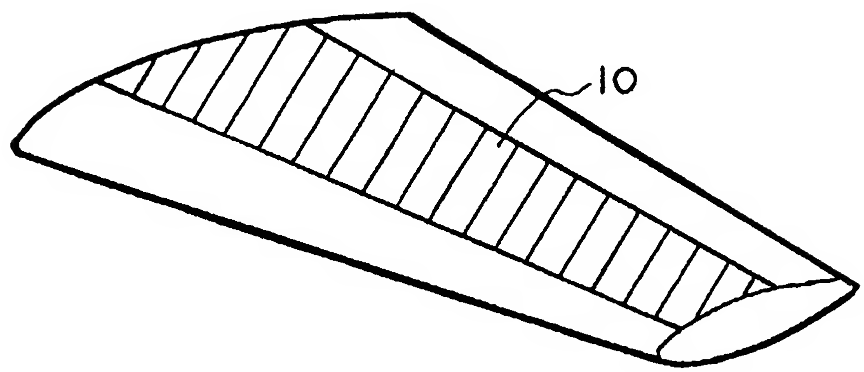


FIG. 19
PRIOR ART

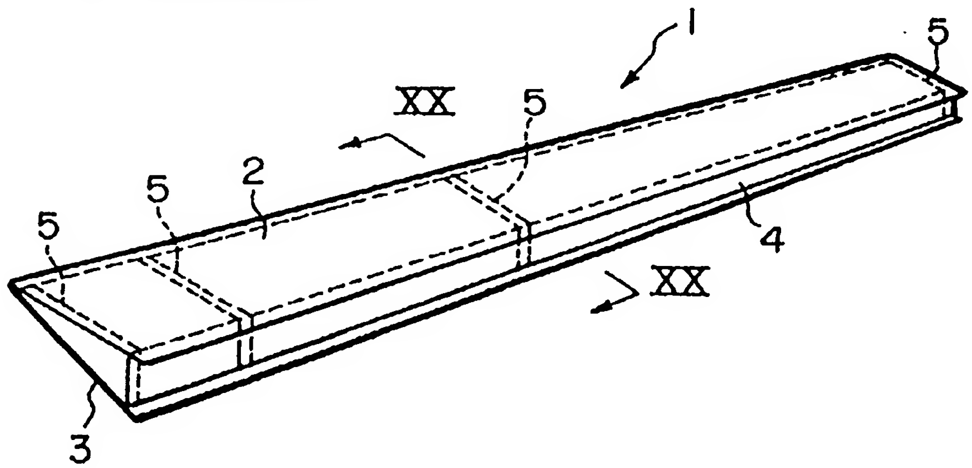


FIG. 20
PRIOR ART

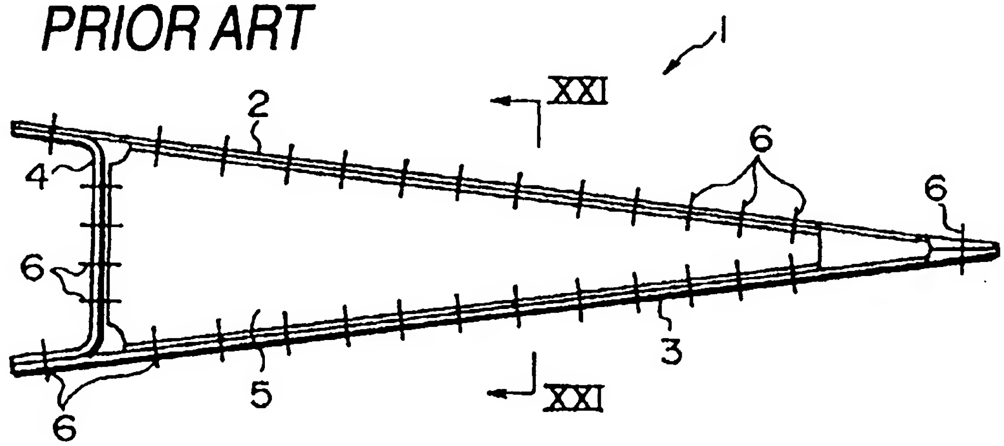


FIG. 21
PRIOR ART

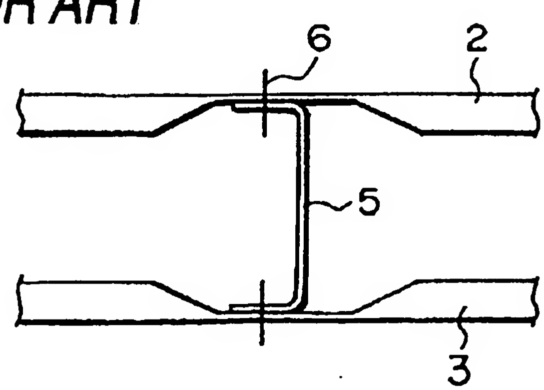


FIG. 22 (a)
PRIOR ART

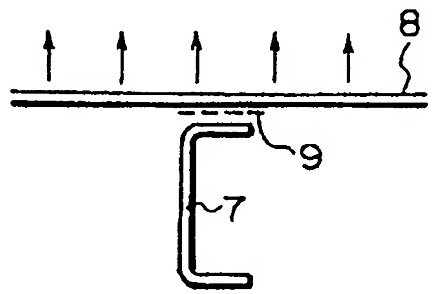


FIG. 22 (b)
PRIOR ART

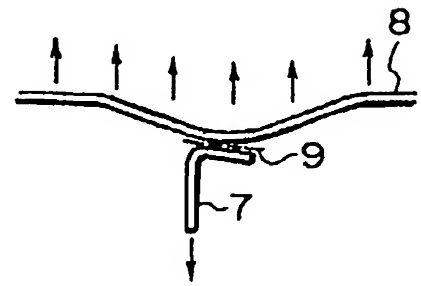


FIG. 23 (a)
PRIOR ART

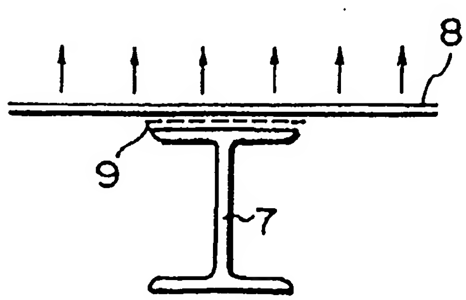
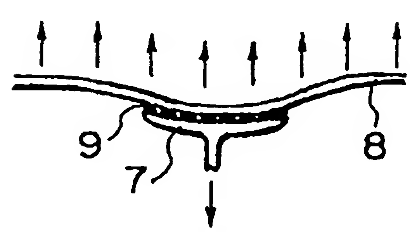


FIG. 23 (b)
PRIOR ART



COMPOSITE AIRFOIL STRUCTURES AND THEIR FORMING METHODS

BACKGROUND OF THE INVENTION

1. Field of the Invention

The present invention pertains to airfoil structures of composite material used in part of an aircraft elevator or wing, for example, as well as to methods of forming such airfoil structures. More particularly, the invention relates to airfoil structures of composite material which are easy to manufacture and provide a high peeling strength as well as to their forming methods.

2. Description of the Related Art

FIGS. 19 to 21 show an elevator surface structure of a related art aircraft using composite materials, in which the aircraft's elevator **1** is constructed by individually forming an upper skin **2**, a lower skin **3**, a spar **4** and ribs **5** using the composite materials and, then, assembling them by use of fastener means **6**, such as bolts and nuts.

The related art aircraft elevator **1** has such problems that it involves a large number of principal constituent components, requiring high manufacturing costs for those components, and because they need to be assembled by using a number of fasteners, assembling component costs are also high. A further problem is that it is necessary to make many holes for fitting the fasteners, resulting in an increase in man-hours required for assembly work.

Box-structure airfoils in which a frame is produced by previously fastening spars and ribs with clips to reduce the man-hours required for assembly with fasteners and upper and lower skins are bonded to the frame are disclosed in European Patent Bulletin Publication No. 485027 and published U.S. Pat. No. 5,216,799, for example.

Although it has been attempted to reduce the number of fasteners and the man-hours required for assembly by assembling the individual skins by bonding in the aforementioned box-structure airfoil, the number of components is not actually reduced because the principal components have the same construction as those of the related art. Further, as it is necessary to previously join the spars and ribs with fasteners and then adjust mating surfaces by machining flange surfaces of the spars and ribs after frame assembly, there arises a problem that man-hour requirements are increased.

Also, structures shown in FIGS. 22 and 23 in which a frame **7** and a skin **8** are bonded by using an adhesive **9** have a low out-of-plane peeling load (peeling load exerted in a direction perpendicular to the skin **8**) compared to fastener assembly, and this would pose a problem related to strength.

Although it is advantageous for improving the peeling strength if the frame **7** has an I-shaped cross section rather than a U-shaped cross section, the frame **7** of the I-shaped cross section entails approximately twice as high manufacturing cost as the frame **7** of the U-shaped cross section, thus developing a problem of increased cost.

SUMMARY OF THE INVENTION

The invention has been made in consideration of these situations. Accordingly, it is an object of the invention to provide airfoil structures of composite material which make it possible to reduce the number of principal constituent components and assembling components to thereby achieve cost reduction, as well as methods of forming such airfoil structures.

A composite material airfoil structure of the invention comprises a composite material skin which forms one of top and bottom surfaces of an airfoil, a second composite material skin which forms the other of the top and bottom

surfaces of the airfoil, a composite material spar having flanges and a web which together form a U-shaped cross section, the spar being attached to at least one terminal portion of the skins, composite material ribs and an elongate projection for adhesive bonding of the spar, the ribs and elongate projection being located between the skins, wherein the ribs and elongate projection are formed integrally with at least one of the skins. In this airfoil structure, it becomes possible to reduce the number of principal constituent components and thereby achieve cost reduction. Furthermore, by bonding the individual flanges of the spar to individual skin members, it becomes possible to obtain a peeling strength equivalent to that achieved when using a spar having an I-shaped cross section even when the spar having the U-shaped cross section is used to achieve cost reduction.

In a composite material airfoil structure of the invention, the ribs and elongate projection are formed integrally with one skin and their extreme ends are bonded to the other skin. In this airfoil structure, the construction of the skin having no ribs or elongate projection is simplified and it becomes possible to reduce manufacturing costs.

In a composite material airfoil structure of the invention, the ribs and elongate projection are formed integrally with each skin and their extreme ends are bonded to one another. In this airfoil structure, dimensions of the ribs and elongate projection rising out of each skin are reduced and it becomes possible to improve the overall strength of each skin one-piece formed with the ribs and elongate projection.

A method of forming a composite material airfoil structure of the invention comprises a molding process in which a first composite material skin which forms one surface of an airfoil and a second composite material skin which has ribs and an elongate projection integrally formed on an inner surface and forms the other surface of the airfoil are separately formed, and a bonding process in which bonding between extreme ends of the ribs and elongate projection and the first skin, bonding between individual flanges of a spar and the individual skins, and bonding between a web of the spar and the elongate projection are simultaneously done by using an adhesive to thereby form a single structure. This method makes it possible to reduce the number of principal constituent components and thereby achieve cost reduction. It also becomes possible to reduce the manufacturing costs as it is not necessary to use fasteners, for instance.

In a method of forming a composite material airfoil structure, the first skin is formed by laminating composite material prepreg on a mold base having the same outer contours as the airfoil, placing a foam core covered with a glue film on an upper surface of the prepreg, laminating again composite material prepreg on top, and hardening an entire assembly by thermosetting operation. Since this method uses the foam core as core material, pressures are uniformly allocated during a hardening process unlike the case in which an anisotropic core material like a honeycomb panel is used. Thus, it becomes possible to reduce deformation during the thermosetting operation.

In a method of forming a composite material airfoil structure, the second skin is formed by laminating composite material prepreg on a mold base having the same outer contours as the airfoil, placing a foam core covered with a glue film on the upper surface of the prepreg, laminating again composite material prepreg on top, placing an intra-projection foam core covered with a glue film on top, laminating again composite material prepreg on top, placing intra-rib foam cores covered with a glue film on top, laminating yet again composite material prepreg on top, and then hardening an entire assembly by thermosetting operation. Since this method uses the foam cores as core material, deformation is reduced even when the second skin and the

ribs and elongate projection are one-piece molded. It also becomes possible to reduce the number of principal constituent components by one-piece molding the second skin with the ribs and elongate projection.

A method of forming a composite material airfoil structure of the invention comprises a molding process in which a first composite material skin which forms one surface of an airfoil and a second composite material skin which has ribs and an elongate projection integrally formed on an inner surface and forms the other surface of the airfoil are separately formed, and a bonding process in which bonding between extreme ends of ribs and elongate projections of the individual skins, bonding between individual flanges of a spar and the individual skins, and bonding between a web of the spar and the elongate projections are simultaneously done by using an adhesive to thereby form a single structure. This method makes it possible to reduce the number of principal constituent components and thereby achieve cost reduction. It also becomes possible to reduce the manufacturing costs as it is not necessary to use fasteners, for instance. Furthermore, since the ribs and elongate projection are integrally formed in halves on each skin, their dimensions rising out of each skin are reduced and it becomes possible to improve the overall strength of each skin one-piece formed with the ribs and elongate projection.

In a method of forming a composite material airfoil structure, each of the first and second skins is formed by laminating composite material prepreg on a mold base having the same outer contours as the airfoil, placing a foam core covered with a glue film on the upper surface of the prepreg, laminating again composite material prepreg on top, placing an intraprojection foam core covered with a glue film on top, laminating again composite material prepreg on top, placing intra-rib foam cores covered with a glue film on top, laminating yet again composite material prepreg on top, and then hardening an entire assembly by thermosetting operation. Since this method uses the foam cores as core material, it becomes possible to reduce deformation during the thermosetting operation even when the ribs and elongate projection are one-piece molded with each skin member.

In a method of forming a composite material airfoil structure, pressure plates are placed at least on end surfaces of the ribs and elongate projection prior to the thermosetting operation, the pressure plates having shapes corresponding to the shapes of mating bond surfaces. This arrangement eliminates the need to adjust the mating surfaces one by one when bonding the two skins to each other and makes it possible to simplify mismatch correction.

In a method of forming a composite material airfoil structure, a pasty thermosetting adhesive is used as the adhesive. As a consequence, a gap, whichever created between bond surfaces, is filled with the adhesive as long as the gap is about 2 mm wide or less, and it becomes possible to prevent a reduction in strength almost completely.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. 1 is a cross-sectional structural diagram of a box-structure airfoil according to a first embodiment of the invention;

FIG. 2 is an exploded perspective diagram of the box-structure airfoil of FIG. 1;

FIG. 3 is a diagram showing a location where the box-structure airfoil of FIG. 1 is applied;

FIG. 4 is a diagram showing a method of forming a skin member;

FIG. 5 is a cross-sectional diagram taken along lines V—V of FIG. 4;

FIG. 6 is a diagram showing locations of molding jigs and a caul plate fitted to skins prior to thermosetting operation;

FIG. 7 is a cross-sectional diagram taken along lines VII—VII of FIG. 6;

FIG. 8 is a diagram showing a skin which is subjected to the thermosetting operation with the molding jigs and caul plate set in position;

FIG. 9 is a diagram showing a method of bonding the two skins and a spar;

FIG. 10 is a cross-sectional diagram taken along lines X—X of FIG. 9;

FIG. 11 is a perspective diagram showing a status in which the airfoil has been set in a bonding jig, which is used when bonding the box-structure airfoil, for carrying out a bonding operation;

FIG. 12 is a vertical cross-sectional diagram taken from FIG. 11;

FIGS. 13*a*–13*b* is a diagram showing stresses exerted on ribs having a trapezoidal cross section;

FIG. 14 is a diagram corresponding to FIG. 9 showing a second embodiment of the invention;

FIG. 15 is a cross-sectional diagram taken along lines XV—XV of FIG. 14;

FIG. 16 is a diagram corresponding to FIG. 1 showing a third embodiment of the invention;

FIG. 17 is a cross-sectional diagram taken along lines XVII—XVII of FIG. 16;

FIG. 18 is a diagram showing a location where a box-structure airfoil of FIG. 16 is applied;

FIG. 19 is a perspective diagram showing the construction of an elevator of a related art aircraft of composite material;

FIG. 20 is an enlarged cross-sectional diagram taken along lines XX—XX of FIG. 19;

FIG. 21 is an enlarged cross-sectional diagram taken along lines XXI—XXI of FIG. 20;

FIGS. 22*a*–22*b* is a diagram showing stresses exerted on adhesively bonded joints between a frame having a U-shaped cross section and a skin; and

FIGS. 23*a*–23*b* is a diagram showing stresses exerted on adhesively bonded joints between a frame having an I-shaped cross section and a skin.

DESCRIPTION OF THE PREFERRED EMBODIMENTS

Modes of carrying out the invention, or embodiments thereof, are described below referring to the drawings.

FIGS. 1 and 2 show a box-structure airfoil using composite materials according to a first embodiment of the invention. This box-structure airfoil 10 is used as an elevator of an aircraft as shown in FIG. 3.

The box-structure airfoil **10** comprises a composite material upper skin **11** forming a top surface of the airfoil, a composite material lower skin **12** forming a bottom surface of the airfoil, and a composite material spar **13** attached to an extreme forward portion of the airfoil as shown in FIGS. 1 and 2, in which four ribs **14**, for instance, and an elongate projection **15** for adhesive bonding of the spar **13** are formed integrally on the inner surface of each skin **11**, **12**. Both skins **11**, **12** and the spar **13** are bonded by a pasty thermosetting adhesive to together form a single structure.

As shown in FIGS. 4 and 5, each of the skins **11**, **12** is formed by laminating composite prepreg **18** like a thermosetting carbon-fiber reinforced plastic (CFRP), for instance, on a steel skin forming jig **17** having a surface corresponding to the outer contours of the airfoil, placing an intra-skin foam core **20** whose outer surface is covered with a glue film **19** like a thermosetting film, for instance, on the upper surface of the composite prepreg **18** laminate, laminating

again composite prepreg 18 on top, placing an intra-projection foam core 21 covered with a glue film 19 on top, laminating again composite prepreg 18 on top, placing intra-rib foam cores 22 covered with a glue film 19 on top, laminating yet again composite prepreg 18 on top, and then hardening the entire assembly by application of heat.

It is advisable to use a product known by the trade mark ROHACELL of Rohm Co., Ltd. of Germany that is made up of polymethacrylic imide which provides excellent heat resistance as the foam cores.

The individual foam cores 20, 21, 22 used are foam cores which have been subjected to postcuring operation (one form of heat treatment) to reduce deformation which may occur while they are molded under heat. The intra-projection foam core 21 and the intra-rib foam cores 22 are formed into a rodlike structure having a trapezoidal cross section as shown in FIGS. 4 and 5. These foam cores 21, 22 improve overall strength as they are combined with the intra-skin foam core 20 and prevent stress concentration.

The skins 11, 12 thus formed are set such that end surfaces of their ribs 14 and elongate projections 15 align face to face, and joined together using the adhesive, as shown in FIGS. 1 and 2.

The spar 13 is made up of flanges 13*a* located on both sides and a web 13*b* which together form a U-shaped cross section as shown in FIGS. 1 and 2. The flanges 13*a* on both sides are bonded to the individual skins 11, 12 by the adhesive. The web 13*b* is bonded to the elongate projections 15 of both skins 11, 12 by the adhesive. With this adhesive bond construction, an adhesive strength equivalent to the adhesive strength achieved by a spar having an I-shaped cross section is obtained by using the spar 13 of the U-shaped cross section.

The spar 13 is produced by laminating composite prepreg like thermosetting CFRP, for instance, shaping the composite prepreg laminate into a U-shaped cross section, and then hardening it by heat.

Next, an airfoil forming method according to the present embodiment is described.

In the manufacture of the box-structure airfoil 10, the upper skin 11, the lower skin 12 and the spar 13 are formed individually.

In forming each skin 11, 12, the composite prepreg 18 like a 180° C.-hardening CFRP, for instance, is laminated on the skin forming jig 17 having a surface corresponding to the outer contours of the airfoil, the intra-skin foam core 20 whose outer surface is covered with the glue film 19 like a 180° C.-hardening film, for instance, is placed on the upper surface of the composite prepreg 18 laminate, and the composite prepreg 18 is laminated again on top as shown in FIGS. 4 and 5. The skins are formed in this manner.

Subsequently, the intra-projection foam core 21 is covered with the glue film 19 and placed on the upper surface, and the composite prepreg 18 is laminated again on top. Consequently, the elongate projection 15 is formed.

Further, the intra-rib foam cores 22 are covered with the glue film 19 and placed on the upper surface, and the composite prepreg 18 is laminated yet again on top. Consequently, the ribs 14 are formed.

On the laminated assembly thus obtained, a jig 23 made of steel, for instance, for keeping a surface to be bonded to the spar 13 in flat shape is placed at the front side of the elongate projection 15, a jig 24 made of CFRP, for instance, for keeping skin bonding surfaces in flat shape is placed on the end surfaces of the ribs 14 and elongate projections 15, and a caul plate 25 made of aluminum, for instance, for forming a trailing edge flat surface is placed at the rear end of the laminated assembly, as shown in FIGS. 6 to 8. The entire assembly is then covered with bleeder cloth 26 and a back pack 27 and bagging is done by sealing a gap between the periphery of the back pack 27 and the skin forming jig 17 with a sealing material 28. Then, each skin 11, 12 is molded under heat at a temperature of 180° C. under a pressure of 3.2 atmospheres in an autoclave.

On the other hand, the spar 13 is formed as follows. Composite prepreg formed of a thermosetting CFRP, for instance, is laminated and a resultant laminate, formed into a U-shaped cross-sectional shape, is molded under heat at a temperature of 180° C. under a pressure of 3.2 atmospheres in the autoclave to produce the spar 13.

When the individual skins 11, 12 and the spar 13 have been formed in the aforementioned manner, a pasty thermosetting adhesive 16 (which cures at 120° C., for example) is applied to bond surfaces of the two skins 11, 12 and those of the spar 13, and the same adhesive 16 is applied to outer surfaces of the two flanges 13*a* and the web 13*b* of the spar 13 as shown in FIGS. 9 and 10. Then, these components are bonded simultaneously to together form a single structure.

FIGS. 11 and 12 show an example of a bonding jig used for bonding operation, in which the bonding jig 30 has a base 31, a pair of face plates 33 mounted on the base 31 by supports 32, together forming a V shape, and wedge-shaped load application blocks 34. The two skins 11, 12 and the spar 13 to which the adhesive 16 has been applied are inserted into a space between the two face plates 33 from above in a temporarily bonded condition and pressed by the load application blocks 34 placed on the spar 13. The skins 11, 12 and the spar 13 are bonded to form the single structure as they are heated at a temperature of 120° C. under pressure exerted by the blocks 34.

Because the ribs 14 and the elongate projection 15 are formed as integral part of each skin 11, 12 as described above, the number of principal constituent components is reduced. Since the assembling components like fasteners are not required as a consequence, it is possible to achieve cost reduction.

Furthermore, since the foam cores 20, 21, 22 having isotropic mechanical properties are used as core material and these foam cores are postcured, it is possible to reduce deformation which may occur while they are molded under heat even when each skin 11, 12 is one-piece molded together with the ribs 14 and the elongate projection 15.

Also, because the jigs 23, 24 and the caul plate 25 are placed on the adhesive bond areas of each skin 11, 12 prior to their molding operation under heat, it is possible to significantly improve the accuracy of their bond surfaces. In particular, it become no longer necessary to adjust mating surfaces of the individual ribs 14 one by one in adhesive assembly operation and this serves to simplify mating surface adjustment.

In addition, since the spar 13 is formed into a U-shaped cross-sectional shape, it can be produced at a lower cost compared to those having an I-shaped cross section. Moreover, since not only the web 13*b* but also the flanges 13*a* on both sides of the spar 13 are bonded to the respective skins 11, 12, it is possible to obtain an adhesive strength equivalent to the adhesive strength achieved by the spars having the I-shaped cross section.

Furthermore, because the individual ribs 14 have a trapezoidal cross section as shown in FIGS. 13(*a*) and 13(*b*), their left and right inclined surfaces can evenly support a peeling load. Due to this trapezoidal cross-sectional shape, combined with the fact that the individual skins 11, 12 has high stiffness and the amount of their deformation is small, it is possible to alleviate stress concentration. Moreover, although the extreme end surfaces of the ribs 14 of the individual skins 11, 12 are bonded face to face, it is possible to obtain a sufficient peeling strength since the peeling load is evenly supported by the left and right inclined surfaces of each rib 14.

FIGS. 14 and 15 are diagrams showing a second embodiment of the invention, in which the upper skin 11 and the lower skin 12 of the first embodiment are replaced by an upper skin 41 and a lower skin 42.

More particularly, the upper skin 41 is constructed of a skin portion alone, while the lower skin 42 is constructed of a skin portion and large-sized ribs 44 and elongate projection 45 which are one-piece formed with the skin portion, as shown in FIGS. 14 and 15.

This embodiment otherwise has the same construction and functional features as the earlier-described first embodiment.

Although the ribs 44 and elongate projection 45 of the lower skin 42 are large-sized compared to those of the first embodiment, the present embodiment has an advantage over the first embodiment in terms of cost when applied to a box-structure airfoil 10 having a relatively small thickness, because it is advantageous in that the upper skin 41 is made into a simple shape.

FIGS. 16 to 18 are diagrams showing a third embodiment of the invention which is so configured that a box-structure airfoil 10 can be used as a horizontal stabilizer of an aircraft.

More particularly, this box-structure airfoil 10 is constructed of an upper skin 51, a lower skin 52, a front spar 53A and a rear spar 53B, in which ribs 54, a front elongate projection 55A for bonding the front spar 53A and a rear elongate projection 55B for bonding the rear spar 53B are formed integrally on the inner surface of each skin 51, 52, as shown in FIG. 16.

This embodiment otherwise has the same construction and functional features as the earlier-described first embodiment.

In the light of the foregoing discussion, it is expected that this embodiment provides the same advantageous effects as the earlier-described first embodiment.

Since ribs and elongate projection are one-piece molded together with a skin member in this invention as thus far described, it is possible to reduce the number of principal constituent components as well as the number of assembling components like fasteners. As a consequence, it is possible to achieve cost reduction. Further, since individual flanges of a spar are bonded to individual skin members, it is possible to obtain a peeling strength equivalent to that achieved when using a spar having an I-shaped cross section even when the spar having a U-shaped cross section is used to achieve cost reduction.

In this invention, the ribs and elongate projection are formed integrally with one skin and their extreme ends are bonded to the other skin. Therefore, the shape of the second skin can be simplified and, in particular, it is possible to achieve cost reduction when the invention is applied to a box-structure airfoil having a small thickness.

In this invention, the ribs and elongate projection are formed integrally with each skin and their extreme ends are bonded to one another. This makes it possible to reduce dimensions of the ribs and elongate projection rising out of each skin even when the invention is applied to a box-structure airfoil having a large thickness. Furthermore, such skins are easy to manufacture and it is possible to improve the overall strength of each skin as it is one-piece formed with the ribs and elongate projection.

In this invention, a box-structure airfoil is manufactured by a molding process in which a spar and individual skin members are separately formed and a bonding process in which bonding between extreme ends of the ribs and elongate projection and the first skin member, bonding between individual flanges of the spar and the individual skins, and bonding between a web of the spar and the elongate projection are simultaneously done by using an adhesive to

thereby form a single structure. This makes it possible to reduce the number of principal constituent components as well as the number of assembling components like fasteners and thereby achieve cost reduction.

In this invention, the first skin is formed by laminating composite material prepreg on a mold base having the same outer contours as the airfoil, placing a foam core covered with a glue film on an upper surface of the prepreg, laminating again composite material prepreg on top, and hardening an entire assembly by thermosetting operation. By using the foam core as core material, pressures are uniformly allocated during a hardening process unlike the case in which an anisotropic core material like a honeycomb panel is used. Furthermore, since the foam core is covered with the glue film, it is possible to improve adhesion between the foam core and the prepreg.

In this invention, the second skin is formed by laminating composite material prepreg on a mold base having the same outer contours as the airfoil, placing a foam core covered with a glue film on the upper surface of the prepreg, laminating again composite material prepreg on top, placing an intra-projection foam core covered with a glue film on top, laminating again composite material prepreg on top, placing intra-rib foam cores covered with a glue film on top, laminating yet again composite material prepreg on top, and then hardening an entire assembly by thermosetting operation. According to this arrangement, the second skin is one-piece molded with the ribs and elongate projection, making it possible to reduce the number of principal constituent components.

In this invention, a box-structure airfoil is manufactured by a molding process in which a spar and individual skins are separately formed, and a bonding process in which bonding between extreme ends of ribs and elongate projections of the individual skins, bonding between individual flanges of the spar and the individual skins, and bonding between a web of the spar and the elongate projections of both skins are simultaneously done by using an adhesive to thereby form a single structure. Since the ribs and elongate projection are integrally formed in halves on each skin, their dimensions rising out of each skin are reduced even when the box-structure airfoil is of a type having a large thickness. Furthermore, the individual skins can be easily formed and it is possible to improve the overall strength of each skin as it is one-piece formed with the ribs and elongate projection.

In this invention, each of the first and second skins is formed by laminating composite material prepreg on a mold base having the same outer contours as the airfoil, placing a foam core covered with a glue film on the upper surface of the prepreg, laminating again composite material prepreg on top, placing an intra-projection foam core covered with a glue film on top, laminating again composite material prepreg on top, placing intra-rib foam cores covered with a glue film on top, laminating yet again composite material prepreg on top, and then hardening an entire assembly by thermosetting operation. According to this arrangement, it is possible to reduce deformation during the thermosetting operation even when the ribs and elongate projection are one-piece molded with each skin.

In this invention, pressure plates are placed at least on end surfaces of the ribs and elongate projection prior to the thermosetting operation, the pressure plates having shapes corresponding to the shapes of mating bond surfaces. This arrangement eliminates the need to adjust the mating surfaces one by one when bonding the two skins to each other and makes it possible to simplify mismatch correction.

Since a pasty thermosetting adhesive is used as the adhesive in this invention, a gap, whichever created between bond surfaces, is filled with the adhesive and this makes it possible to prevent a reduction in strength almost completely.

While presently preferred embodiments of the present invention have been shown and described in detail, it will be understood that these disclosures for the purpose of illustration and that various changes and modifications can be made therein without departing from the spirit and scope of the invention as set forth in the appended claims.

What is claimed is:

1. An airfoil structure made from a composite material, having, an upper skin made from said composite material for forming an upper contour of said airfoil structure and for extending in a chord length direction, a lower skin made from said composite material for forming a lower contour of said airfoil structure and for extending in said chord length direction, and a rib interposed between said upper and lower skins and arranged in said chord length direction for supporting thereof, comprising:

a spar interposed between said upper and lower skins at either one of edge portions thereof in said chord length direction and extended in a span-wise direction;

a web formed in said spar and being approximately perpendicular to said both skins for making a space therebetween;

a flange formed on both edge portions of said web for supporting said upper or lower skin so as to adequately and continuously mate with said upper or lower contour; and

a protrusion integrally formed on said upper or lower skin for adhesively bonding said spar and said rib so as to effectively and economically maintain a sufficient strength against a peeling stress applied thereof.

2. A box-shaped wing airfoil structure having, an upper skin made of a composite material for forming an upper contour surface thereof, a lower skin made of said composite material for forming a lower contour surface thereof, and a spar made of said composite material and interposed between said upper and lower skins to extend in a spanwise direction of the wing airfoil structure, comprising:

flanges formed at both upper and lower edge portions of said spar and bonded to a lower surface of a front edge portion of said upper skin and to an upper surface of a front edge portion of said lower skin, respectively;

a web formed between rear edges of said flanges of said spar to provide the spar with a U-shaped cross-section opening forward and to define a rear surface of the spar;

a first elongated projection of said composite material, integrally formed with along a lower surface of the front edge portion of said upper skin and bonded to said rear surface of the spar;

a second elongated projection of said composite material, integrally formed with and along an upper surface of the front edge portion of said lower skin and bonded to said rear surface of the spar;

a plurality of first ribs of said composite material, formed integrally with both said upper skin and said first elongate projection and extending rearward from the first elongate projection said first ribs being disposed with intervals in said spanwise direction;

a plurality of second ribs formed integrally with both said lower skin and said second elongate projection and extending rearward from the second elongate projection, said second ribs being disposed with intervals in said spanwise direction; and

said first elongate projection and said first ribs being bonded to said second elongate projection and said second ribs in a mutually confronting disposition;

whereby precise and easy assembly of the box-shaped wing airfoil structure at a low cost is enabled with a strong resistant strength against a peeling force applied thereof.

3. The box-shaped wing airfoil structure according to claim 2, wherein:

said first and second elongate projections have a trapezoidal cross-sectional shape.

4. The box-shaped wing airfoil structure according to claim 2, wherein:

said first and second ribs have a trapezoidal cross-sectional shape.

5. A box-shaped wing airfoil structure having, an upper skin made of a composite material for forming an upper contour surface thereof, a lower skin made of said composite material for forming a lower contour surface thereof, and a spar made of said composite material and interposed between said upper and lower skins to extend in a spanwise direction of the wing airfoil structure, comprising:

flanges formed at both upper and lower edge portions of said spar and bonded to a lower surface of a front edge portion of said upper skin and to an upper surface of a front edge portion of said lower skin, respectively;

a web formed between rear edges of said flanges of said spar to provide the spar with a U-shaped cross-section opening forward and to define a rear surface of the spar;

an elongate projection of said composite material, integrally formed with and along an upper surface of the front edge portion of said lower skin and bonded to said rear surface of the spar;

a plurality of ribs of said composite material, formed integrally with both said lower skin and said elongate projection and extending rearward from the second elongate projection, said ribs being disposed with intervals in said spanwise direction; and

said elongate projection and said ribs being bonded to a lower surface of said upper skin in a confronting disposition;

whereby precise and easy assembly of the box-shaped wing airfoil structure at a low cost is enabled with a strong resistant strength against a peeling forced applied thereof.

6. The box-shaped wing airfoil structure according to claim 5, wherein:

said elongate projection has a trapezoidal cross-sectional shape.

7. The box-shaped wing airfoil structure according to claim 5, wherein:

said ribs has a trapezoidal cross-sectional shape.

* * * * *

US005224670A

United States Patent [19]

Padden

[11] **Patent Number: 5,224,670**

[45] **Date of Patent: Jul. 6, 1993**

[54] **COMPOSITE FOCUSED LOAD CONTROL SURFACE**

[75] Inventor: **Vincent T. Padden, Brightwaters, N.Y.**

[73] Assignee: **Grumman Aerospace Corporation, Bethpage, N.Y.**

[21] Appl. No.: **759,231**

[22] Filed: **Sep. 13, 1991**

[51] **Int. Cl.**[5] **B64C 1/26**

[52] **U.S. Cl.** **244/123; 244/213**

[58] **Field of Search** .... 244/117 R, 119, 213, 244/214, 215, 131, 132, 133, 123; 52/806

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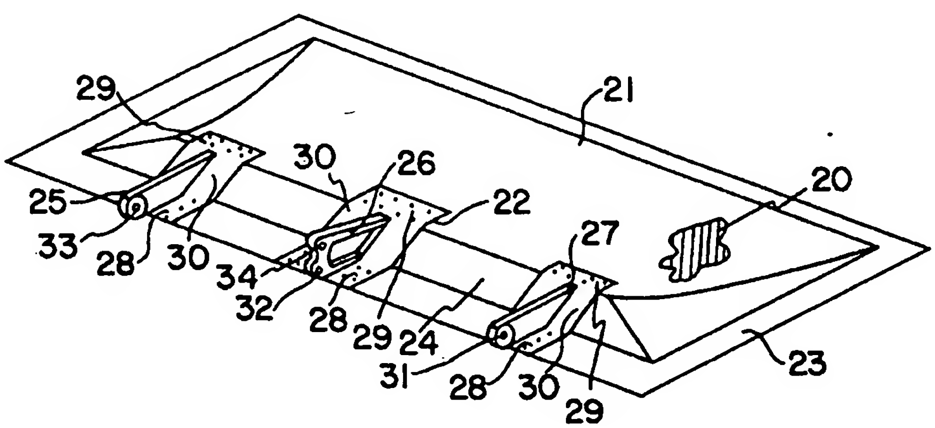
Primary Examiner—Galen Barefoot

Attorney, Agent, or Firm—Bacon & Thomas

[57] **ABSTRACT**

A graphite epoxy spoiler for an aircraft wing is manufactured by estimating the shear force to which the spoiler will be subject, and forming a fitting attachment surface and fitting designed to eliminate a separate shear attachment between the spoiler structure and the center attachment/drive fitting. This permits use of a one piece co-cured honeycomb sandwich construction for the spoiler.

24 Claims, 7 Drawing Sheets



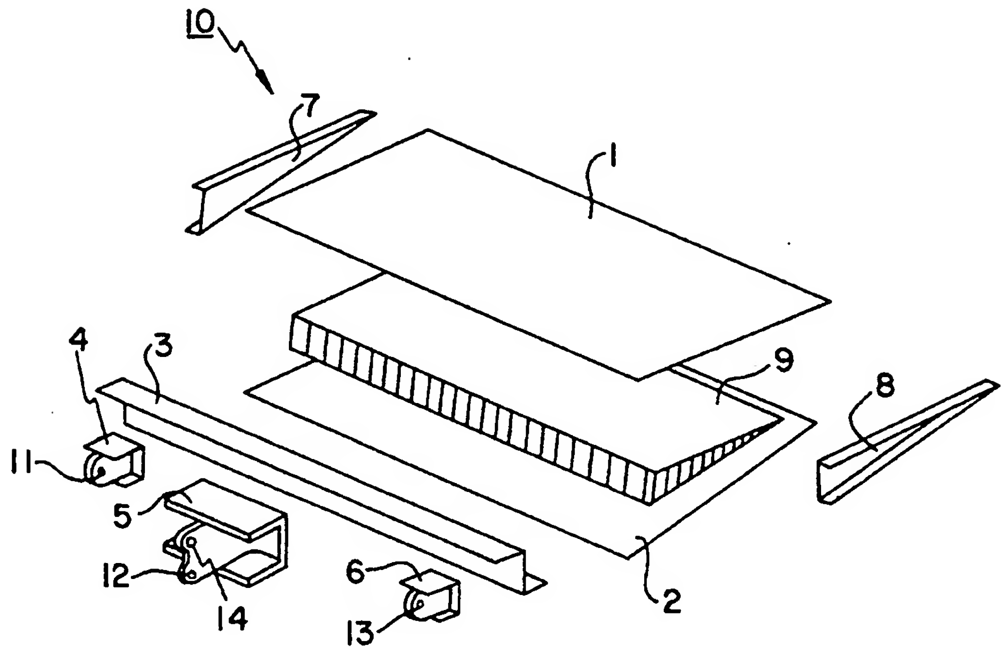

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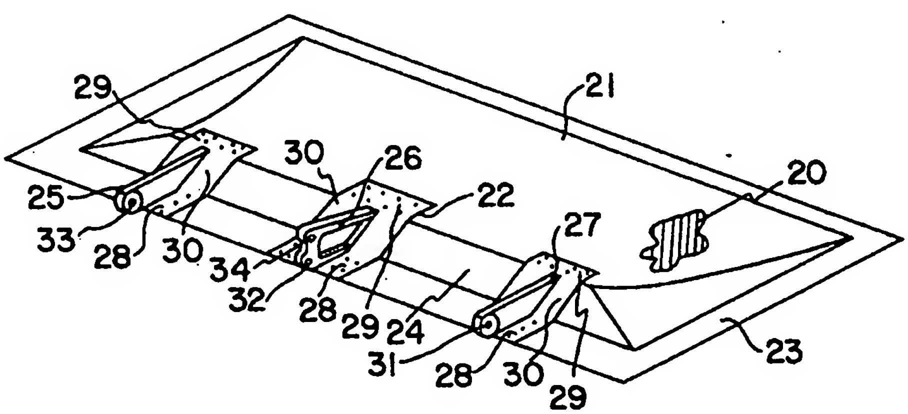
FIG.1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23

FIG.2

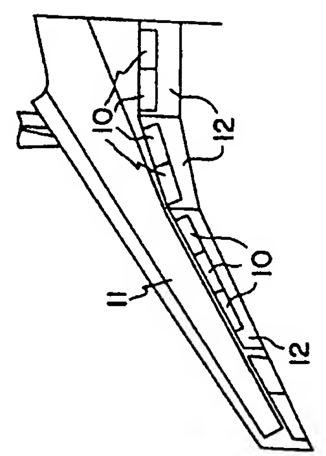


FIG.3a

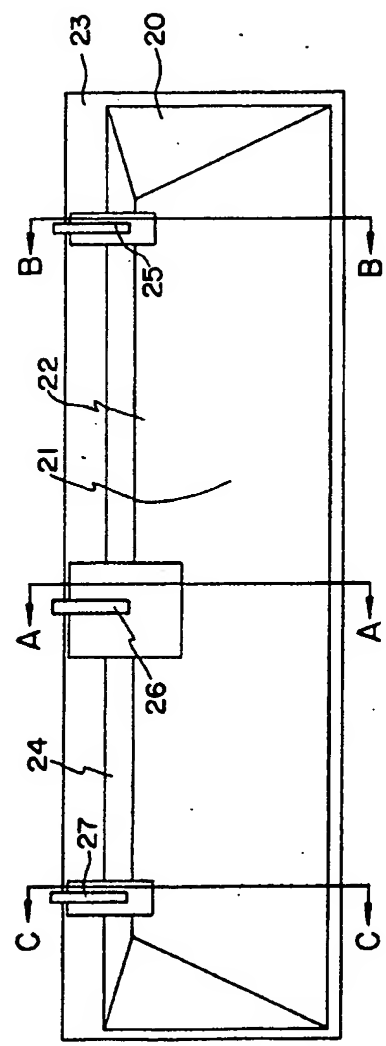


FIG.3b

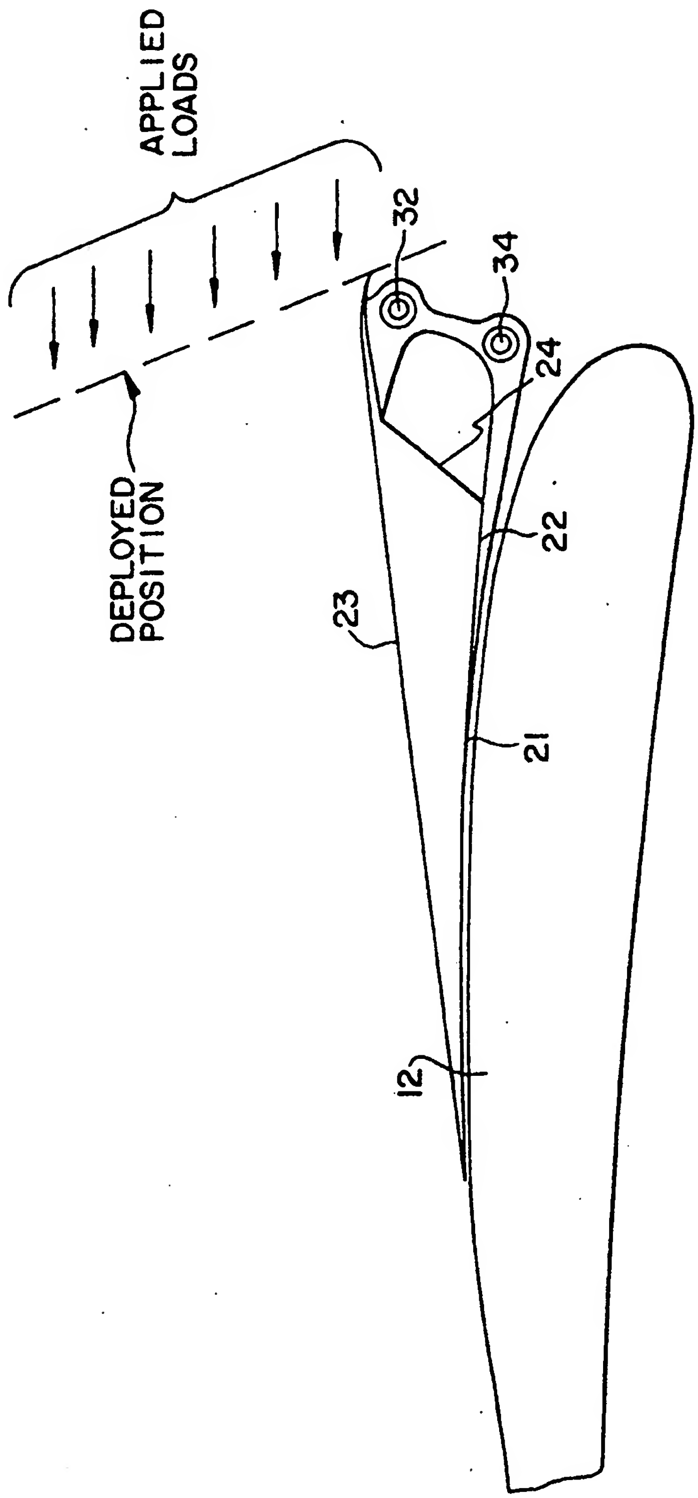


FIG. 4

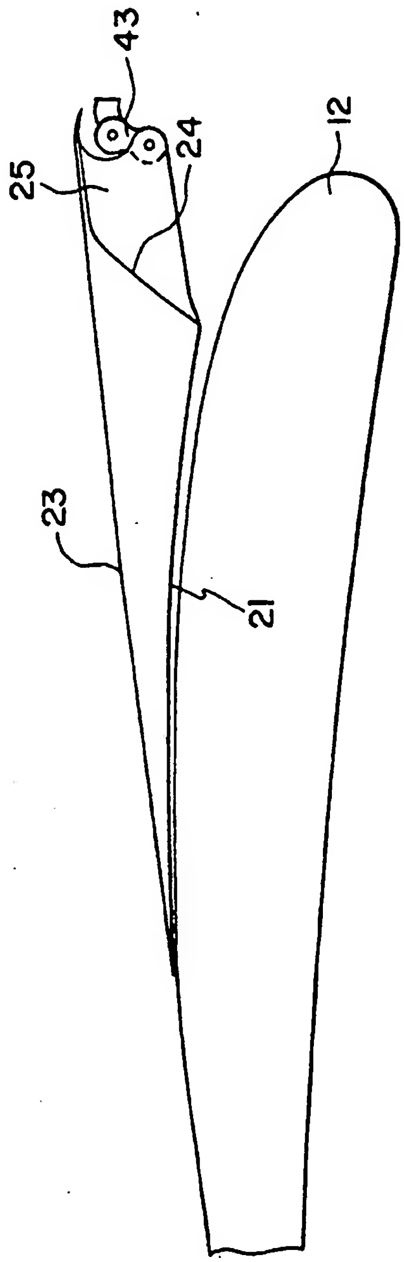


FIG.5

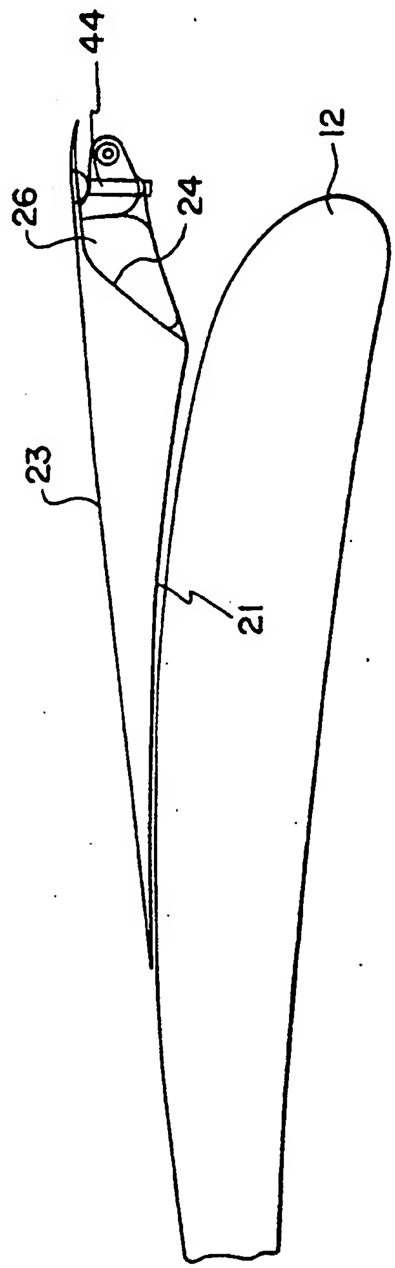


FIG. 6

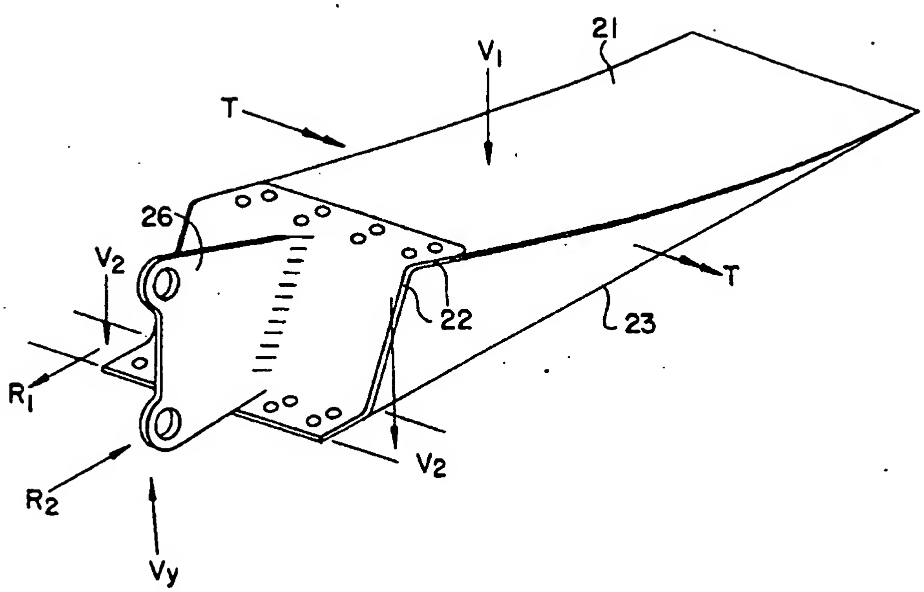
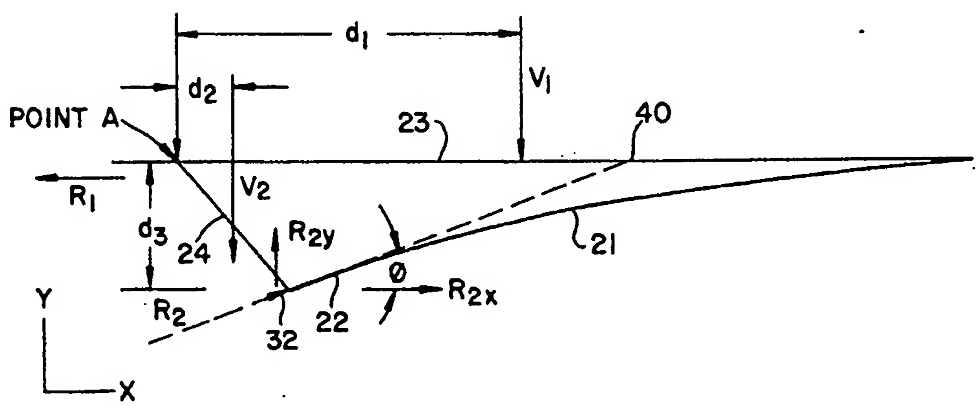

21
V1
T
26
V2
22
T
23
R1
V2
R2
Vy

FIG.7


d1
d2
V1
POINT A
23
40
R1
V2
d3
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R2y
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Ø
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R2
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22
X

FIG.8

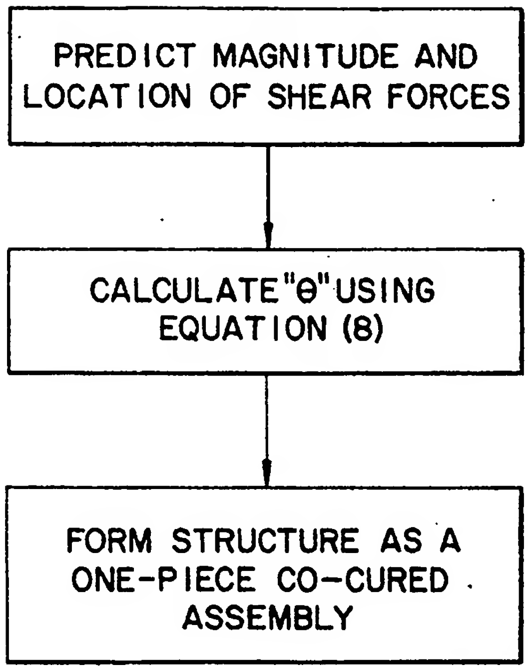
PREDICT MAGNITUDE AND LOCATION OF SHEAR FORCES
CALCULATE "θ" USING EQUATION (8)
FORM STRUCTURE AS A ONE-PIECE CO-CURED ASSEMBLY

FIG. 9

COMPOSITE FOCUSED LOAD CONTROL SURFACE

BACKGROUND OF THE INVENTION

1. Field of the Invention

This invention relates to the manufacture of composite load bearing structures, and in particular to the manufacture of an aircraft wing spoiler of honeycomb sandwich construction. The invention also relates to a design configuration for composite load bearing structures in general and in particular to a design configuration for an aircraft wing spoiler of honeycomb sandwich construction.

2. Description of Related Art

Because of the weight savings offered by advanced composite materials, especially epoxy graphite, and the resulting improvements in performance, modern aircraft have become increasingly reliant on such materials. The F-16 was the first military aircraft to use graphite epoxy structures in production, and the technology has advanced to the point where many aircraft, such as the F-18, use graphite composites extensively, in applications ranging up to 26% of the aircraft's structural weight. Future military aircraft designs are projected to include as much as 40–50% of their structural weight in advanced composites.

The commercial and general aviation industries have also utilized advanced composite technology, in conventional transport aircraft, the resulting weight savings providing significant increases in payload and consequent decreases in fuel consumption. Advanced composites have already been certified by the FAA for secondary structural (not critical to flight safety) components, and transport aircraft have been designed with advanced composite secondary structures including fairings, control surfaces, and the like.

A disadvantage of the graphite epoxy control surface, however, is that the cost to weight ratio is relatively high in comparison with metals such as aluminum. The reason for the high cost is that structural requirements for graphite epoxy control surfaces currently necessitate a concentration of graphite material piles and, in many configurations, the use of multiple pre-cured parts in order to provide spars and other structural elements necessary to accommodate fitting attachments. Construction of control surfaces having a large number of graphite material piles and pre-cured parts requires multiple curing cycles in the autoclave, greatly increasing manufacturing time and costs.

FIG. 1 is an exploded view of a current graphite epoxy spoiler design for a transport aircraft. The spoiler **10** is a special form of control surface which is located on the upper surface of the trailing edge of the wing **11** as shown in FIG. 3*a*, and which deflects upward from flaps **12** under pilot command in order to provide roll control and braking functions. The principal load on the spoiler when the spoiler is extended is therefore a shear force resulting from the airstream at the top of the wing which is normal to the spoiler when the spoiler is in its operative position.

The current design utilizes a honeycomb sandwich core construction, and requires six pieces, not including the three attachment fittings. The six pieces include respective pre-cured upper and lower covers **1** and **2**, a pre-cured spar **3** for providing shear attachment of mounting fittings **4–6** to the control surface, and pre-cured closure ribs **7** and **8** which enclose the honeycomb core **9**. In order to construct this type of spoiler, the various parts are separately pre-cured and then bonded together to form the spoiler assembly. The mounting fittings **4–6** are then attached to spar **3**. The mounting hardware defines three hinge apertures **11–13** and an aperture **14** for attaching an actuator control rod.

This design, while structurally and aerodynamically acceptable, suffers from high manufacturing costs due to the relatively large number of graphite plies and curing steps to form the final assembly.

SUMMARY OF THE INVENTION

In view of the above-described disadvantages of current designs, it is an objective of the invention to reduce costs by providing a method of manufacturing a load-bearing structure subjected during use to shear forces, but which eliminates the need for a direct shear attachment of the structure's mounting fittings to the surface of the structure.

It is a second objective of the invention to reduce costs by providing an advanced composite control surface subjected during use to shear forces and which would conventionally require a multiple part construction, but which is instead a co-cured one piece assembly.

It is a third objective of the invention to provide a method of manufacturing an aircraft spoiler assembly which does not require a direct shear attachment of the spoiler actuator mounting fitting to the spoiler's control surface.

It is a fourth objective of the invention to provide a method of manufacturing an advanced composite aircraft control surface, and in particular a graphite epoxy control surface, which eliminates the need for a multistep curing process employing multiple pre-cured pieces.

These objectives are accomplished by providing a co-cured one piece assembly in which the attachment fitting of the assembly's control surface is caused to converge at the focus of the applied loads on the attachment fitting. This eliminates the need for direct shear attachment of the hinge/actuator drive fitting to the control surface because the shear load is carried as a component of the axial load at the fitting attachment surface.

As a result of the inventive design, it is possible to provide graphite epoxy construction of aircraft control surfaces which is competitive in cost with traditional aluminum construction. This approach allows for a less complicated graphite structure around the main attachment fitting because no vertical shear attachment is required between the attachment fitting and the bonded graphite panel, and thus a separate pre-cured forward spar on the control surface is not required. Attachment fitting complexity is also reduced using the inventive focused load concept because less fasteners are required along with a smaller fitting.

In a particularly preferred embodiment of the invention, a center attachment fitting for an aircraft spoiler is provided with a center spoiler hinge point and drive actuator attachment point. The core of the spoiler includes a one-piece co-cured honeycomb sandwich construction with an integral front spar supported on the wing by two hinge fittings and a center actuator drive fitting. The spoiler is driven by one mid-span hinge/actuator connected to a single hinge/actuator drive fitting which is designed such that the transverse shear is re-

acted by a component of the axial force in the lower cover at the hinge/actuator drive fitting. The inboard fitting is a link fitting used to uncouple wing and spoiler strong axis bending. The outboard fitting has a pivot line normal to the hinge line to eliminate side loads. All side loads are reacted at the center hinge/actuator fitting.

Focusing of the applied loads is accomplished by causing an extension or tangent of the lower cover fitting attachment flange to converge at a focus point on the upper cover to form an angle of load convergence selected to cause the resulting structure to function as an integral unit or "truss."

According to the preferred method of manufacture, therefore, the above spoiler, including a composite load bearing surface of honeycomb sandwich construction, is formed as a one piece co-cured assembly by first predicting the shear forces to which the control surface will be subject, and then forming the lower fitting attachment flange such that an extension thereof converges on the upper surface at the focus point of the predicted shear forces.

In an especially advantageous embodiment of the invention, the integral spar is at an angle of approximately forty-five degrees to allow the lower cover to drape completely over the entire core.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. 1 is an exploded perspective view of a current spoiler design.

FIG. 2 is a perspective view of a one-piece co-cured spoiler design having an integral front spar according to a preferred embodiment of the invention.

FIG. 3(*a*) is a top view of an aircraft wing showing the location of spoilers of the type illustrated in FIGS. 1-3(*a*).

FIG. 3(*b*) is a top view of the preferred spoiler of FIG. 2.

FIG. 4 is a cross-sectional side view of the preferred spoiler taken along line A—A in FIG. 3(*b*).

FIG. 5 is a side view of the preferred spoiler taken along line B—B in FIG. 3.

FIG. 6 is a side view of the preferred spoiler taken along line C—C in FIG. 3.

FIG. 7 is a perspective view illustrating the applied loads to the preferred spoiler hinge/actuator drive fitting.

FIG. 8 is a schematic diagram further illustrating the geometry of the applied loads to the preferred hinge/actuator drive.

FIG. 9 is a flowchart illustrating a preferred method of manufacturing a load bearing structure according to a preferred embodiment of the invention.

DETAILED DESCRIPTION OF THE PREFERRED EMBODIMENTS

FIGS. 2-6 show a one piece co-cured spoiler structure constructed according to the principles of a preferred embodiment of the invention. FIGS. 1, 2, and 3(*b*) are shown in inverted position so that the lower cover is depicted as facing up for clarity.

The preferred spoiler, as shown in FIG. 2, is made up of a honeycomb core 20, a lower surface 21 including a hinge attachment area 22, an upper surface 23, and an integral front spar 24. The spoiler is attached to the wing by hinge fittings 25 and 26 and by an hinge/actuator drive fitting 27. Each of the attachment fittings includes mounting surfaces or flanges 28 and 29 and an intermediate surface 30. The fittings also include hinge attachment apertures 31-33 and actuator attachment aperture 34 similar to respective apertures 11-14 of the conventional spoiler design. Flange 29 and surface 21 form an angle determined by a line extending from the flange to the location of the focal point of the predicted shear force on the control surface when the spoiler is extended, as will be described in detail below.

The preferred material for the spoiler is an advanced composite material such as graphite epoxy while the material for all hinge fittings is preferably aluminum. However, it will be appreciated that the invention is not intended to be limited to a particular material for either the fittings or the spoiler itself. In the case of a graphite epoxy spoiler with aluminum fittings, all aluminum parts are preferably isolated from the graphite epoxy using a fiberglass isolation ply.

Construction of the spoiler employs the previously mentioned, co-cured molding process and the focused load concept to minimize weight, part bonding costs, and assembly costs, including minimizing the costs of hole drilling, fastener insulation, fit up, and shimming. Use of a co-cured molding process is possible because the design of the spoiler reduces complexity of the center hinge/actuator drive fitting attachment points. The upper and lower cover design utilizes tape and fabric carbon epoxy while the edges of the spoiler are preferably thickened to resist handling damage. Preferably, the integral spar is at forty-five degrees, as shown, to permit the lower cover to drape completely over the entire spar.

The core may be in the form of a Nomex honeycomb, or a similar advanced composite honeycomb construction. The bonded spoiler assemblies are mechanically assembled to the aluminum hinge fittings using flush bolt and nuts through the upper cover and, by way of example, blind Composi-loks for the connections through the lower cover and integral spar.

This simple construction, using well-known curing techniques, is made possible through the use of an attachment flange geometry which eliminates a shear attachment on the hinge/actuator drive fitting, as follows:

Referring to FIGS. 7 and 8, which shows a section of the spoiler at the hinge/actuator drive fitting 27, it is initially noted that the spoiler loads may be represented as shear force V_1 and $2V_2$ and opposing coupling forces R_1 and R_2. Force R_1 is provided by the actuator and is opposed by the parallel or horizontal hinge coupling force component R_{2x} of R_2. The shear force V_1 and $2V_2$ is opposed by the transverse or vertical component R_{2y} of hinge coupling force R_2.

The scalar sum of all forces on the fitting in the X direction is therefore simply the sum of R_1 and R_{2x}, i.e.:

$$\Sigma F_x = -R_1 + R_{2x} = 0 \quad (1)$$

The scalar sum of the vertical or Y direction forces is

$$\Sigma F_y = -V_1 + R_{2y} = 0 \quad (2)$$

At the hinge attachment point 32, the shear force V_y actually has two components, the first component V_1 being the aerodynamic force on the lower cover as shown in FIG. 7. The second component V_2 of shear force V_r is distributed over the spar 24. Thus, $\Sigma V = V_1 + 2V_2$, while

$$\Sigma F_y = R_{2y} - V_1 - 2V_2 = 0 \quad (3)$$

Σ Fy is set to zero because the system is effectively static when in operation, and is preferably determined at the maximum load to which the spoiler will be subject for all flight and ground modes.

In FIG. 8, the scalar sum of the moments acting about point A is therefore given by the sum of the products of the respective unconstrained forces R_{2x}, R_{2y}, V_1 and V_2 and their respective distances to point A, d_3, $2d_2$, d_1, and d_2 (one half the spar depth), with the addition of a torsion T caused by air loads on the remaining spoiler surface adjacent to the hinge/actuator area. This sum is set to zero about point A, as follows:

$$2V_2d_2 + V_1d_1 + T - R_{2y}(2d_2) - R_{2x}(d_3) = 0 \quad (4)$$

As is apparent from FIG. 8, the relative magnitudes of the coupling forces R_{2x} and R_{2y}, which are the reactive forces on the hinge opposed to actuator coupling force R_1, are determined by the shape of the fitting, and thus by the angle between attachment flange 22 and the X axis or horizontal direction defined by the lower cover. In order to eliminate the shear force on the hinge fitting therefore, it is simply necessary to specify θ in a manner which satisfies the initial condition given by equation (4).

Since

$$\theta = \arctan \frac{R_{2y}}{R_{2x}} \quad (5)$$

$$R_{2x} = 2V_2\left(\frac{d_2}{d_3}\right) + V_1\left(\frac{d_1}{d_3}\right) + \frac{T}{d_3} - R_{2y}\left(\frac{d_2}{d_3}\right) \quad (6)$$

and

$$R_{2y} = V_1 + 2V_2. \quad (7)$$

then

$$\tan\theta = \frac{V_1 + 2V_2}{V_1\frac{d_1}{d_3} + \frac{T}{d_3} - 2(V_1 + 2V_2)\left(\frac{d_2}{d_3}\right)} \quad (8)$$

Equations 6 and 7 are solutions of equations 4 and 3, respectively, in terms of R_{2x} and R_{2y}, θ can thus be defined solely in terms of predictable shear forces acting on the spoiler and the spoiler geometry. For loading conditions where the pressure load distribution on the control surface varies, the determination of the distance to the focal point (d_1), would be based on a weighted average loading condition and location of the center of pressure. Moments produced by shear forces not located on the focus would be reacted by vertical forces on the fitting flanges. Under normal conditions these forces would be small and considered as a secondary effect in the design.

Once the angle of the attachment surface is defined, the hinge/actuator drive fitting 27's inner surface flange is designed to match the angle. In addition inboard fitting 25 includes a slotted pivot 3 perpendicular to the hinge line which uncouples the spoiler and wing strong axis bending, while in the spoiler deplayed condition after bending the outboard fitting 26 has a pivot line 44 normal to the hinge line to eliminate side loads so that all side loads are reacted at the center hinge/actuator fitting 27, as was assumed in the calculations used to obtain the geometry of fitting 27.

As a result of the above geometry, it is possible to manufacture a graphite epoxy aircraft spoiler, and other load bearing composite structures, using a single curing step as follows, with reference to FIG. 9:

First, the focii and magnitude of shear forces V_1 and V_2 must be determined using conventional internal loads analysis, so that θ can be calculated according to equation 8. Once θ is obtained, the spoiler is formed with integral lower and upper covers and co-cured as a one-piece assembly. Finally, the attachment hardware may be added to complete the spoiler assembly.

Prediction of the shear forces may be accomplished in a conventional manner, using computer modeling and/or wind tunnel tests of various spoiler configurations. In addition, the curing process by which the spoiler is formed will be readily apparent to those skilled in the art in view of the above description.

Having thus described a specific example of the invention in terms of a graphite epoxy spoiler, it will nevertheless be appreciated that the principles of the invention are not to be limited to aircraft spoiler designs, but rather may have application in a variety of structures requiring shear attachments. In addition, while graphite epoxy is a preferred material, the principles of the invention may be of use in connection with other materials that are used in contexts in which a shear interface is required. In fact, it is anticipated that numerous variations of the invention will occur to those skilled in the art. Therefore, it is intended that the invention not be limited to the specific embodiment described, but rather that it be limited solely by the appended claims.

I claim:

1. A method of manufacturing a load bearing structure which during use is subjected to a shear force, comprising the steps of:

(a) forming a core having a lower surface and an upper surface, and a spar connected to said lower and upper surfaces to form a continuous cover for said core;

(b) determining a focus point of applied loads on said fitting; and

(c) forming in said lower surface a fitting attachment flange for attaching a fitting to said structure and which is connected to said spar such that a tangent of said attachment flange converges with said upper surface at the focus point of applied loads on said fitting.

2. A method as claimed in claim 1, further comprising the step of curing said core, lower surface, upper surface, and spar simultaneously to obtain a one piece cocured structure.

3. A method as claimed in claim 1, wherein said core, spar, and lower and upper surfaces are formed as a co-cured one piece structure having a honeycomb core sandwiched by said lower and upper surfaces.

4. A method as claimed in claim 3, wherein said attachment surface and spar are formed as a single continuous surface.

5. A method as claimed in claim 4, wherein said spar is formed at a non-zero angle relative to a principal plane of said lower surface.

6. A method as claimed in claim 1, wherein said step of forming said fitting attachment flange comprises the step of causing said tangent to converge at said upper surface at a convergence angle θ arranged to eliminate

a vertical shear connection between said spar and said attachment flange.

7. A method as claimed in claim 6, wherein said angle of convergence θ of said tangent and said lower surface is defined by the equation

$$\tan\theta = \frac{V_1 + 2V_2}{V_1\left(\frac{d_1}{d_3}\right) + \frac{T}{d_3} - 2(V_1 + 2V_2)\left(\frac{d_2}{d_3}\right)},$$

where V_2 is a net shear force on said spar, V_1 is a net shear force on said upper surface, d_2 is one half a horizontal component of a spar depth, said component being parallel to a principal plane of said upper surface, and d_3 is a vertical component of the spar depth.

8. A load bearing structure manufactured according to the method of claim 1.

9. A load bearing structure which during use is subjected to a shear force, comprising:

a core; an upper surface; a lower surface including a fitting attachment flange for attaching a fitting to said structure; and a fitting attachment spar connecting the lower surface and the fitting attachment surface, wherein a tangent of said fitting attachment surface and said upper surface converge at a focal point of applied loads on said fitting.

10. A structure as claimed in claim 9, wherein said core, spar, lower surface, and upper surface form a co-cured one piece structure.

11. A structure as claimed in claim 9, wherein said core is a honeycomb core sandwiched by said lower and upper surfaces.

12. A structure as claimed in claim 11, wherein said lower surface and spar are parts of a single continuous surface.

13. A structure as claimed in claim 12, wherein said spar forms a surface oriented at a non-zero angle with respect to said upper surface.

14. A structure as claimed in claim 9, wherein an angle θ between said tangent of said attachment surface and said upper surface is given by the following equation:

$$\tan\theta = \frac{V_1 + 2V_2}{V_1\left(\frac{d_1}{d_3}\right) + \frac{T}{d_3} - 2(V_1 + 2V_2)\left(\frac{d_2}{d_3}\right)},$$

where V_2 is a net shear force on said spar, V_1 is a net shear force on said upper surface, d_2 is one half a horizontal component of a spar depth, said component being parallel to a principal plane of said upper surface, and d_3 is a vertical component of the spar depth.

15. A structure as claimed in claim 9, wherein said structure is an aircraft wing spoiler.

16. A structure as claimed in claim 9, wherein said structure is formed of a composite material.

17. A structure as claimed in claim 16, wherein said composite material is graphite epoxy.

18. A structure as claimed in claim 9, further comprising means including at least one hinge fitting for pivotably attaching the structure to a second structure.

19. A structure as claimed in claim 18, wherein said hinge fitting is made of aluminum.

20. A structure as claimed in claim 18, wherein a number of said fittings is three.

21. A structure as claimed in claim 20, wherein the actuator/hinge attachment fitting includes a first portion arranged to be mounted on said second structure, a second portion oriented to be parallel to said attachment surface, and a third portion connecting said first and second portions, said third portion being oriented to be parallel to a surface of said spar.

22. A structure as claimed in claim 20, wherein each of said fittings includes means defining an aperture for mounting said structure on a pivot to form a hinge, and wherein one of said fittings further includes means defining an aperture for mounting an actuator to cause said structure to pivot about said hinge.

23. A structure as claimed in claim 22, wherein said first structure is a spoiler and said second structure is an aircraft wing having inboard and outboard portions.

24. A structure as claimed in claim 23, wherein an outboard one of said fittings has means including a pivot line extending normal to an axis of said hinge for eliminating side loads.

* * * * *

IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re Application of:

WOLFGANG BILLINGER, et al. Group Art Unit: 3644

Serial No. 10/053,666 Examiner: S. Holzen

Filed: January 24, 2002

For: DEVICE FOR CONNECTING MOVABLE PARTS WITH STRUCTURAL ELEMENTS OF AIRPLANES AND THE LIKE

DECLARATION OF WOLFGANG BILLINGER

Commissioner of Patents
P.O. Box 1450
Alexandria, VA 22313-1450

Sir:

I, Wolfgang Billinger, do hereby declare the following:

1. I reside in Ried, Austria, am currently employed by Fischer Advanced Composite Components AG (FACC AG) at Fischerstr:9, P.O. Box 192, A-4910 Ried, Austria, and am one of the named inventors for the above-identified application.

2. My educational background is set forth on my Curriculum Vitae ("CV") attached hereto as Attachment A. As indicated, I have a Master's degree in engineering from the technical university of Vienna. I have also been published extensively in the field of composite materials.

3. My professional experience, which is also set forth on the attached CV, Attachment A, includes over 14 years of experience as a senior engineer with FACC AG, at least six of which have been spent

in the development and testing of connecting devices for the securing of structural components and movable parts of aircraft. Throughout my tenure with FACC AG, I have gained extensive experience in calculating monolithic and sandwich structures of state of the art composite materials, as well as metallic airplane structures. As a result of my education and experience, my professional contemporaries and colleagues consider me an expert in composite structures and materials, specifically as such composite technology relates to the aviation field.

4. Based upon my education and experience, I have an opinion regarding what the level of ordinary skill in the aviation field as related to composite technology generally would be. In most cases, such persons would have at least a bachelor's degree in mechanical engineering, and would have at least several years of experience in mechanical engineering.

5. Drawing upon my expertise in this industry, I understand the level of ordinary skill in the art at the time the above-captioned application was filed (January 24, 2002) and can offer my expert opinion as to how persons of ordinary skill in the art would perceive and respond to relevant art in the field of composite technology, particularly as it pertains to the aviation field.

6. In the aviation field, a fitting suitable for high load application, such as that acting on the movable parts used to assist the aircraft in taking off, landing, and moving, must be especially stable and not prone to separation from the movable part due to

shearing forces. Such shearing forces are caused by the high temperature deviations faced in aviation and the differences in temperature coefficients between synthetic components and metal fittings. However, as representatively shown by U.S. patents to Roeseler et al. (4,213,587) and Arena (5,098,043) as discussed in the specification, persons of ordinary skill in the art attempting to solve these problems did not contemplate a fitting made of synthetic composite material according to a resin transfer molding method, and including a carbon fabric as a reinforcing element, nor did they contemplate a securing of the fitting to the movable part by gluing. Instead, what was known by persons of ordinary skill was reliance upon metal fasteners, often a high number of screws or rivets, for secure connection of *metal* fittings to movable parts.

7. I have reviewed the prior art considered by the Patent Examiner, specifically U.S. Patent No. 4,966,802 to Hertzberg and EP 0 532 016 to Padden.

8. Padden describes a spoiler for an aircraft wing which is disclosed as being made of a composite material. However, there is nothing to suggest the use of a synthetic material *for the fitting* (which instead is made of aluminum), nor is there anything to suggest the formation of the fitting from the same composite material as the movable part so that both parts share a common thermal expansion coefficient; again, a fitting made of synthetic composite material according to a resin transfer molding method, whether joined to the movable part by gluing or formed integrally

therewith, was considered by persons of ordinary skill in the field to be structurally inadequate for the high-load application being claimed by my invention.

9. Hertzberg describes composites made of fiber reinforced resin elements joined by adhesive *for joining sheets or panels of material and their reinforcing members*. These sheets or panels, when adhered together, are resistant to the delamination problems associated with prior resin composites and therefore represent an improvement over prior resin composites. However, Hertzberg does not contemplate the use of fiber reinforced resin elements in the formation of *fittings* to secure movable and structural parts.

10. Persons skilled in the aviation field seeking to design a fitting for high load application between movable and structural parts of an aircraft would look to the art of metal fittings. Such persons would not look to laminated panel construction such as shown in the U.S. patent of Hertzberg (4,966,802), in seeking to design a fitting for the purpose and application claimed in the subject patent application.

11. During my six years of developing and testing connecting devices for the securing of structural components and movable parts of aircraft, I have never seen a fitting made of synthetic composite material according to a resin transfer molding method, and including a carbon fabric as a reinforcing element, with the fitting being secured to the movable aircraft part by gluing. Nor am I aware of anyone in the aviation industry who considered the use of a fitting

of such construction for this application.

12. When a fitting of the claimed type was introduced to the industry in 2002, the response of experts was high interest with some disbelief. Such a fitting was universally considered to be inadequate to withstand the known shearing stresses, as was the use of glue to connect movable and structural aircraft components. Further, in this technology field, namely high-load aircraft applications, bolting and gluing are not considered equivalent by persons of ordinary skill therein.

13. It is my opinion that the subject matter claimed in the above-identified application would not have been obvious to one of ordinary skill in the art based upon the prior art cited and considered by the Patent Examiner, or any other prior art fitting construction and design known to me.

14. I further declare that all statements made herein of my own knowledge are true and that all statements made on information and belief are believed to be true; and further that these statements were made with the knowledge that willful false statements and the like so made are punishable by fine or imprisonment or both, under section 1001 of Title 18 of the United States Code; and that such willful false statements may jeopardize the validity of the application or any patent issuing thereon.

7. 6. 2004
Date

Dipl. Ing. W. Billinger
Wolfgang Billinger

Attachment A: CV of Wolfgang Billinger

Best Available Copy

RESUME

STATUS: May, 2004

NAME: Dipl. Ing. WOLFGANG BILLINGER

COMPANY ADDRESS:
FISCHER ADVANCED COMPOSITE COMPONENTS AG
FISCHERSTR: 9/P.O. BOX 192
A-4910 RIED/AUSTRIA

COMPANY PHONE NUMBER: 0043 7752 616 201

COMPANY FAX NUMBER: 0043 7752 616 8201

HOME ADDRESS:
PETERSKIRCHEN 17
A-4743 PETERSKIRCHEN / AUSTRIA

DATE OF BIRTH: FEBRUARY 02, 1962

NATIONALITY: AUSTRIA

A) EDUCATION:

1968 – 1972 primary school

1972 – 1980 secondary school

1980 school leaving examination of secondary school

October 1980 - September 1981 military service in the Austrian army

1981 - 1986
technical university of Vienna, study for mechanical engineering;
special lectures of light weight structures and finite elements
dissertation at the institute for lightweight structures and aircraft construction

December 1986
graduate engineer, released by the technical university of Vienna

B) WORK EXPERIENCE

1987 - 1990
Project engineer at **Wintersteiger GmbH** (Ried) for woodworking machines (division engineering). Stress Engineer for division sports and division seedmech of Wintersteiger.

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Since 1990
Stress Engineer at **Fischer Advanced Composite Components (FACC) Gmbh.**
for FRP - Components for the US and European Aircraft Industry.

Since 2000
R&D Manager FACC AG

C) SPECIAL EXPERIENCE

Composite Technology

Experience in calculating monolithic structures (MD11 - flap hinge fairings, MD90 - strake, ...) as well as sandwich structures (A320 - hatrack, MD11 - flap hinge fairings, B717 - ceiling panel, sidewall panel, bagrack, A340 – thrust reverser blocker door, ..) , consisting of all state of the art composite materials with duroplastic resin systems.
Experience in calculating metallic airplane structures (A320 - main landing gear door hinge, B717 - bearings and brackets, B717 - z-bridge and intercostals, A340 – trust reverser blocker door fittings, ..) A340 – Spoiler, A340 – blocker door, RTM – Center Fitting
New production methods (liquid mouldings) and simulation of them. M&PE activities like qualification of new materials.

Signature:

Dipl.-Ing. W. Billinger

Wolfgang Billinger
Mai 2004

PATENT
ATTY. DOCKET NO.: P67552US0

IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re Application of:

WOLFGANG BILLINGER, et al.

Group Art Unit: 3644

Serial No. 10/053,666

Examiner: S. Holzen

Filed: January 24, 2002

For: DEVICE FOR CONNECTING MOVABLE PARTS WITH STRUCTURAL ELEMENTS OF AIRPLANES AND THE LIKE

DECLARATION OF HELMUT KAUFMANN & RUDOLF GRADINGER

Commissioner of Patents
P.O. Box 1450
Alexandria, VA 22313-1450

Sir:

We, Helmut Kaufmann and Rudolf Gradinger, do hereby declare the following:

1. We, Helmut Kaufman and Rudolf Gradinger, are both currently employed by the ARC Leichtmetallkompetenzzentrum Ranshofen GmbH (hereafter "LKR"), an R&D company in Austria focused on light metal technology. LKR projects range from alloying to component design, mainly with respect to aluminum and magnesium alloys. However, in the case of design and stress engineering, parts made out of steel, titanium, polymers or ceramics may be treated as well.

2. I, Helmut Kaufman, received my M.S. in mechanical engineering and my Ph.D. in material science from the Montanuniversität Leoben (Austria) in 1987 and 1992, respectively. After serving for one year as a visiting scientist at the Materials

Science Department of the Massachussetts Institute of Technology (MIT) in the United States in 1989, I joined Austria Metall AG (AMAG) where I conducted research work in casting technology. In 1994, I moved to Ube Europe GmbH in Düsseldorf (Germany) where my work focussed on Squeeze Casting and Semi-Solid Casting. Upon returning to Austria in 1997, I took the position of head of the Light Metal Competence Center of LKR in Ranshofen. In 2000, when LKR became a wholly owned subsidiary company of the Austrian Research Centers GmbH, I was appointed Managing Director of LKR.

3. I, Rudolf Gradinger, received my M.S. in mechanical engineering from the Vienna University of Technology (Austria) in 1997. In that same year I joined the Light Metal Competence Center of LKR in Ranshofen and started my research work in light weight design. In 2001, I became the head of the joining technology working group at LKR and, in 2003, expanded my responsibilities to include simulation. In 2004, I merged the working groups of joining technology, simulation and new applications into a new Light Weight Design Group of which I am the head. Also in 2004, I held a lecture at Wels College of Engineering on the topic of simulation and computer aided engineering (CAE) in Materials Engineering.

4. As a result of our collective education and experience, it is our opinion that persons of ordinary skill in the aviation field as related to composite technology would have at least a bachelor's degree in mechanical engineering, and would have at least several years of experience in mechanical engineering.

5. Drawing upon our expertise in this industry and our understanding of the level of ordinary skill in the art at the time the above-captioned application was filed (January 24, 2002), we can offer our expert opinion as to how persons of ordinary skill in the art perceived the use of composite technology as it pertains to the aviation field.

6. Prior to 2004 it was conventional to make aircraft spoiler hinges of aluminum or titanium alloys, mainly forgings. In addition, what was known by persons of ordinary skill was reliance upon metal fasteners, such as screws or rivets, for secure connection of the metal fittings to the movable parts.

7. In 2004, we came to know of the design for a carbon fiber reinforced plastic/resin transfer molding (CFRP/RTM) center hinge fitting, i.e., aircraft spoiler hinge, developed by Mr. Wolfgang Billinger ("the Billinger design") at Fischer Advanced Composite Components AG. We have reviewed the corresponding subject matter of Mr. Billinger's captioned U.S. patent application, namely Serial No. 10/053,666, which to our view sets forth the CFRP/RTM aircraft spoiler hinge according to his design.

8. Upon hearing of the Billinger design, we were initially surprised by this innovation as previously we had not been aware of any RTM approach for a hinge or fitting design for use with aircraft spoilers.

9. Unlike known methods of prepreg manufacture which employ an autoclave to solidify the prepreg material, the RTM process offers

more degrees of freedom in aligning the fibers along the main stresses and thus is the first appropriate composite manufacturing method for bulky structurally loaded parts.

10. Using the Billinger design, weight savings on the order of 50 kg per ~~spoiler~~ aircraft can be achieved without increased cost, as compared with conventional metal fittings. This fact is of great significance and, provided the technical requirements are fulfilled, establishes the cost benefit of the RTM approach as compared to any light weight metal design. (To obtain an analogous weight saving using metal, a comparable light metal design could be made out of titanium, but this would be accompanied by an unacceptable increase in cost.)

11. In aircraft design, one of the areas in which major weight savings are obtained is related to the joining technique used between the hinge and the spoiler. The replacement of screws by adhesive bonding according to the Billinger design therefore provides further weight savings and, unexpectedly, sufficient strength to withstand the known shearing stresses to which movable aircraft parts such as spoilers, landing flaps and control surfaces are subject.

12. In addition, the use of composite material for both the fitting and the spoiler according to the Billinger design results in reduced thermal stresses at the interface between these components due to the similarity in their thermal expansion coefficients. This is an improvement over conventional metal fittings where the thermal

stresses between the metal and the CFRP at the bearing of the center hinge fitting have to be considered and accommodated.

13. It is thus our opinion that a fitting made of synthetic composite material according to the RTM method, whether joined to the movable part by gluing or formed integrally therewith, is innovative as the same would not have been considered by persons of ordinary skill in the field to be structurally adequate for the high-load hinge application to which the Billinger design is directed.

14. We further declare that all statements made herein of our own knowledge are true and that all statements made on information and belief are believed to be true; and further that these statements were made with the knowledge that willful false statements and the like so made are punishable by fine or imprisonment or both, under section 1001 of Title 18 of the United States Code; and that such willful false statements may jeopardize the validity of the application or any patent issuing thereon.

11.4.2005
Date

H. Kaufmann
Dr. Helmut Kaufmann

11.4.2005
Date

Rudolf Grad
Dipl.-Ing. Rudolf Gradinger

PATENT
ATTY. DOCKET NO.: P67552US0

IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re Application of:

WOLFGANG BILLINGER, et al. Group Art Unit: 3644

Serial No. 10/053,666 Examiner: S. Holzen

Filed: January 24, 2002

For: DEVICE FOR CONNECTING MOVABLE PARTS WITH STRUCTURAL ELEMENTS OF AIRPLANES AND THE LIKE

DECLARATION OF HELMUT KAUFMANN

Commissioner of Patents
P.O. Box 1450
Alexandria, VA 22313-1450

Sir:

I, Helmut Kaufmann, do hereby declare the following:

1. I am currently employed by the ARC Leichtmetallkompetenzzentrum Ranshofen GmbH (hereafter "LKR"), an R&D company in Austria focused on light metal technology. LKR projects range from alloying to component design, mainly with respect to aluminum and magnesium alloys. However, in the case of design and stress engineering, parts made out of steel, titanium, polymers or ceramics may be treated as well.

2. I received my M.S. in mechanical engineering and my Ph.D. in material science from the Montanuniversität Leoben (Austria) in 1987 and 1992, respectively. After serving for one year as a visiting scientist at the Materials Science Department of the Massachussetts Institute of Technology (MIT) in the United States in

1989, I joined Austria Metall AG (AMAG) where I conducted research work in casting technology. In 1994, I moved to Ube Europe GmbH in Düsseldorf (Germany) where my work focussed on Squeeze Casting and Semi-Solid Casting. Upon returning to Austria in 1997, I took the position of head of the Light Metal Competence Center of LKR in Ranshofen. In 2000, when LKR became a wholly owned subsidiary company of the Austrian Research Centers GmbH, I was appointed Managing Director of LKR.

3. As a result of my education and experience, it is my opinion that persons of ordinary skill in the aviation field as related to composite technology would have at least a bachelor's degree in mechanical engineering, and would have at least several years of experience in mechanical engineering.

4. Drawing upon my expertise in this industry and my understanding of the level of ordinary skill in the art at the time the above-captioned application was filed (January 24, 2002), I can offer my expert opinion as to how persons of ordinary skill in the art perceived the use of composite technology as it pertains to the aviation field, and can also offer my expert opinion as to structures resulting from the resin transfer molding (RTM) process.

5. Prior to 2004 it was conventional to make aircraft spoiler hinges of aluminum or titanium alloys, mainly forgings. In addition, what was known by persons of ordinary skill was reliance upon metal fasteners, such as screws or rivets, for secure connection of the metal fittings to the movable parts.

6. In 2004, I became aware of the design for a carbon fiber reinforced plastic/resin transfer molding (CFRP/RTM) center hinge fitting, i.e., aircraft spoiler hinge, developed by Mr. Wolfgang Billinger at Fischer Advanced Composite Components AG. I have reviewed the corresponding subject matter of Mr. Billinger's captioned U.S. patent application, namely Serial No. 10/053,666 (hereinafter "the Billinger application"), which to my view sets forth the CFRP/RTM aircraft spoiler hinge according to his design.

7. I have also reviewed the prior art considered by the Patent Examiner, specifically U.S. Patent No. 6,234,423 to Hirahara et al. (hereinafter "Hirahara"), in connection with the Billinger application.

8. Hirahara shows a typical moveable surface of an aircraft, such as an elevator, where the spar 13, consisting of flanges 13a and a web 13b, is an integral part of the movable surface formed by stiffened skins 11 and 12. The spar 13 has six hinges attached, which I have marked with the letter "H" in the attached drawing from Hirahara, that are used to connect the spar to the aircraft structure. According to conventional wisdom and practice, to obtain sufficient structural strength these hinges are made from forged alloys of aluminum, high strength steel or titanium. To the best of my understanding upon review thereof, Hirahara does not address the hinges at all, but only methods of forming the box structure of the movable surface.

9. The spar 13, as constructed according to Hirahara, could

not be produced as a single piece composite part together with the fitting, constituted by the six hinges "H", using known state-of-the-art technology because of the significant differences in wall thicknesses between hinges (thicknesses larger than 0.5 inch, and most likely 1.0 inch) and the web and flanges of the movable part (thicknesses less than 0.2 inch, and most likely 0.1 inches), and the mechanical properties needed in order to acceptably replace a metal fitting made, for example, of forged aluminum alloy material or other higher properties material.

10. The technology according to the Billinger application is the only way to produce fittings having the required mechanical properties, i.e., properties equivalent to high end metallic alloys, using composite materials such as carbon-fiber reinforced materials, when the fittings will be used in parts which have significant wall thickness differentiation, as is the case with aircraft fittings and associated movable parts.

11. Resins used in fiber reinforced systems such as RTM are modified with additives in order to attain mechanical properties similar to those exhibited by forged versions made from alloys of aluminum, high strength steel or titanium. These additives generally modify the flow characteristics of the resin, changing their structure. Complicated geometries, like geometries having significant wall thickness differentiation, cannot be produced with conventional resin systems.

12. It is thus my opinion that a *fitting* made of synthetic

composite material according to the RTM method, whether joined to the movable part by gluing or formed integrally therewith, is structurally different from the structure and process disclosed by Hirahara and further is not obvious to one of ordinary skill in the art in view of the disclosure in Hirahara, especially since Hirahara does not speak to the fitting, i.e., the hinges, at all. Even if one were to make the hinges in Hirahara of the same composite material as that disclosed for the skins and spar 13, such theoretical hinges would lack sufficient structural strength for their intended use and would certainly lack the structural strength realized by the fittings made of synthetic composite material according to the RTM method as claimed by the Billinger application.

13. I further declare that all statements made herein of my own knowledge are true and that all statements made on information and belief are believed to be true; and further that these statements were made with the knowledge that willful false statements and the like so made are punishable by fine or imprisonment or both, under section 1001 of Title 18 of the United States Code; and that such willful false statements may jeopardize the validity of the application or any patent issuing thereon.

31.7.2007
Date

H. Kaufmann
Dr. Helmut Kaufmann

FIG. 1

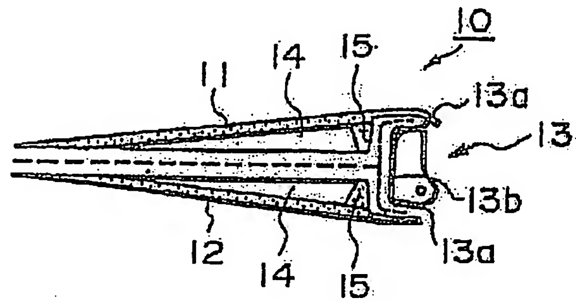
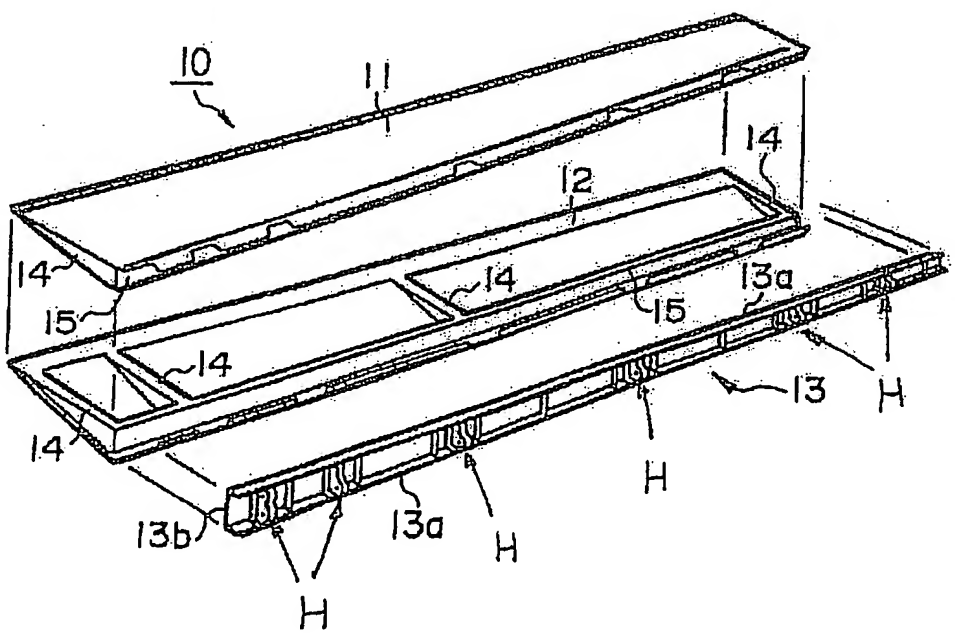

10
11
14
15
13a
13
13b
12
14
15
13a

FIG. 2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11
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12
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15
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13a
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13
H
14
H
13b
13a
H
H

XI. RELATED PROCEEDINGS APPENDIX

There are no known related proceedings at this time.